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Journal

of

THE ROYAL AERONAUTICAL SOCIETY

NOVEMBER 1953

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THE ROYAL AERONAUTICAL SOCIETY

Scholarships and Awards

The awards administered by the Society fall into three main groups:—

(*a*) Scholarships and Educational Grants, (*b*) Medals awarded for achievement in the profession of aeronautics, (*c*) Medals and prizes awarded for papers read before, or published by, the Society. In addition to the named awards there are special premiums attached to the Wilbur Wright Memorial Lecture and to the British Commonwealth and Empire Lecture. Also, there is an annual sum of £250 from which premium awards are made to authors of those papers (including Technical Notes) published in the JOURNAL which are considered to be of particular merit.

SCHOLARSHIPS

The Charter Scholarship—Awarded annually, or at such other times as the Council may decide, to assist a student wishing to undertake advanced work or study in aeronautics. An extension for a second year is looked upon favourably by the Council if it is clear that the student and the work will benefit. The scholarships are awarded only to those who can satisfy the Council that they have the necessary qualifications to benefit and whose proposals for work or further studies are acceptable to the Council. Normally the work should lead to a higher degree or post-graduate diploma.

The Geoffrey de Havilland Memorial Scholarship—Awarded annually, at the discretion of the Council, to assist those engaged in aeronautical research, either theoretical or practical, to undertake advanced study in the field in which they are engaged. The scholarship may be extended for a second year at the discretion of the Council. Scholarships are awarded only to those who can satisfy the Council that they have the necessary qualifications to benefit and whose proposals for work or further studies are acceptable to the Council. Applications should either be graduates or have reached graduate standard.

The Busk Studentship in Aeronautics—Awarded annually at the discretion of the Council to enable the holder to engage in research or preparation for research in aeronautics of the kind which Edward Tashmaker Busk made his own, namely Stability, Meteorology and Gust Research, Aeronautical Instruments, Sighting and Appliances for Bomb Throwing, the calculations and mathematical problems relating to these subjects and generally, any scientific work in connection with any branch of art or trade which conduces to progress in, or the use of flying. The Studentship is awarded for one year but may be extended at the discretion of the Council. It is open to a British Subject less than 25 years of age.

ACHIEVEMENT IN AERONAUTICS

Society's Gold Medal—The highest honour which the Society can confer for work of an outstanding nature in Aeronautics.

Society's Silver Medal—Awarded for work of an outstanding nature in Aeronautics.

Society's Bronze Medal—Awarded for work leading to an advance in Aeronautics.

British Gold Medal—Awarded for outstanding practical achievement leading to advancement in Aeronautics.

British Silver Medal—Awarded for practical achievement leading to advancement in Aeronautics.

Wakefield Gold Medal—Awarded annually, at the discretion of the Council, to the designer or inventor of any apparatus tending towards safety in flying.

R. P. Alston Memorial Prize—Awarded for practical achievement associated with the flight testing of aircraft.

AWARDS FOR PAPERS

The George Taylor (of Australia) Gold Medal—Awarded annually, at the discretion of the Council, for the most valuable contribution read before, or received by, the Society on Aircraft Design, Manufacture or Operation.

Simms Gold Medal—Awarded annually, at the discretion of the Council, for the most valuable contribution read before, or received by, the Society on any subject allied to aeronautics, such as, structures, meteorology, metrology.

Herbert Akroyd Stuart Memorial Prize—Awarded, at the discretion of the Council, for the most valuable contribution read before, or received by, the Society on Applied Thermodynamics.

Edward Busk Memorial Prize—Awarded annually, at the discretion of the Council, for the most valuable contribution read before, or received by, the Society on Applied Aerodynamics.

Pilcher Memorial Prize—Awarded annually, at the discretion of the Council, for the most valuable paper read by a Graduate or Student during the previous year at any meeting of the Society or its Branches.

Usborne Memorial Prize—Awarded annually, at the discretion of the Council, for the best contribution to the Society's publications written by a Graduate or Student on some subject of a technical nature in connection with Aeronautics.

Orville Wright Prize—This prize is offered annually for the best contribution received for publication in the AERONAUTICAL QUARTERLY of the Society on some subject of a technical nature in connection with Aeronautics.

Branch Prize—The Council offer an annual prize for the best paper read before the Branches during the previous lecture session. The Prize is open to any member of the Society or of any Branch.

November 1953

Journal of

THE ROYAL AERONAUTICAL SOCIETY

WITH WHICH IS INCORPORATED THE INSTITUTION OF AERONAUTICAL ENGINEERS

PUBLISHED MONTHLY AT

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ADDITIONS TO THE LIBRARY REPORTS

All communications for publication in, and advertisements for, the Journal should be addressed to the Editor, Joan Bradbrooke, A.R.Ae.S.; all communications on general matters affecting the Society should be addressed to the Secretary, A. M. Ballantyne, T.D., B.Sc., Ph.D., A.F.R.Ae.S.

None of the papers or paragraphs must be taken as expressing the opinion of the Council of the Royal Aeronautical Society unless stated otherwise.

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Secretary's News Letter

November 1953

THE Fourth Anglo-American Aeronautical Conference is now with the others—in the pages of history. Most people appeared to enjoy it and have been very complimentary about it. I have come to the conclusion that Conferences are very like small-boat sailing—pleasurable in retrospect. It was fine to meet friends of previous years and to make new friends.

To many people the thanks of the Society are due: to the Staff of the Society who worked long, laborious days (and nights); to Shell-Mex and B.P. who provided the brief cases, with no demur when the number requested changed suddenly from 600 to 750. Although I could extend this list I shall only name additionally the aircraft firms who made the Visits successful—The Bristol Aeroplane Co. Ltd., Rolls-Royce Ltd., Vickers-Armstrongs Ltd. (Weybridge), Handley-Page Ltd., and Saunders-Roe Ltd. These gave very great satisfaction to the American visitors.

It has already been announced that the Fifth Anglo-American Aeronautical Conference will be held in June 1955 in America. This gives us all something to which we can look forward.

One thing struck strange in my ears: I cannot become used to the idea of calling an American by the familiar "Mac."

The Council has given approval for the founding of the 25th Branch of the Society, at Boscombe Down. The inaugural meeting of the Branch will be on 24th November 1953, with an inaugural address by Sir Frederick Handley Page. The Branch is to be wished every success.

I have received a letter from Squadron Leader Miller, the Secretary of the Singapore Branch, and he reports good progress; the Branch among the other activities has held a very successful Garden Party.

The notice regarding the 50th Anniversary Dinner which is to be held at The Dorchester on 17th December to commemorate the first powered controlled flight has been circulated and I hope that very many people will attend this, the concluding function of a not uneventful year. The tables at the Dinner will seat ten and if anybody wishes to sit beside a friend, will he let me know and I shall try to arrange it. As has been announced, the Dinner will be followed by the first showing of the historical film which has been made by the Shell Film Unit, in collaboration with the Society. Much "carding" and "discarding" has gone into the making of the film—"Powered Flight—the Story of the Century," and I am sure it will be of great interest.

A. M. Ballantyne

Secretary

NOTICES

An annual sum of £250 is available for premium awards, usually 15 guineas each, for papers published in the Journal. Members and non-members of the Society are invited to submit papers on any aspect of aeronautics

Commemorative Dinner—17th December

Particulars of the Dinner which is being held on 17th December 1953 at the Dorchester by the Society, in conjunction with the Royal Aero Club, to commemorate the Fiftieth Anniversary of Powered Controlled Flight, have already been sent to members.

At present there is no restriction on the number of tickets (price £3 10s. each, inclusive of wines) each member may have but early application is advisable. Members are reminded that the closing date for applications for tickets is the 1st December 1953.

Associate Fellowship Examination

The next Associate Fellowship Examinations will be held in the offices of the Society on Tuesday, Wednesday and Thursday, 8th, 9th and 10th December. Particulars of time tables and general arrangements have been sent to candidates. The Library will be closed.

Graduates' and Students' Section—R.A.E. Visit

Because of the heavy demand from members wishing to visit the Royal Aircraft Establishment in August a similar visit has been arranged for *Wednesday 2nd December* 1953, from 10.30 a.m. to 4.30 p.m.. Lunch will be available. Visitors will be able to see the wind tunnels, structural and engine test rigs, and the flight test department.

Applications should be sent to the Hon. Visits Secretary, D. J. W. Richards, 18 Geneva Road, Kingston-on-Thames, Surrey. Members are asked to state their grade of membership of the Society and that they are British Subjects. Applications must be sent as soon as possible as numbers are strictly limited.

Graduates' and Students' Secton—Imperial College

A visit to the Imperial College of Science and Technology has been arranged for Wednesday 27th January 1954, at 6.30 p.m. Visitors will be able to see the wind tunnels, structural test rigs and so on. Supper will be available at the College at Moderate charges. Applications, stating whether dining or not, should be sent to the Hon. Visits Secretary, D. W. J. Richards, 18 Geneva Road, Kingston-on-Thames, Surrey.

Members' New Appointments

T. H. Blake (*Associate Fellow*) has been appointed Inspector-in-Charge, Aeronautical Inspection Directorate, at the de Havilland Aircraft Company.

L. G. Fairhurst (*Fellow*) has been appointed Chief Engineer of British Messier Limited. He will continue to hold a similar post with Rotol Limited.

J. P. Ford (*Associate*) has been appointed Managing Director of Associated British Oil Engines (Export) Ltd. He is also Managing Director of Brush Export Ltd., and of National Oil Engines (Export) Ltd.

E. V. Hall (*Associate Fellow*) has been appointed Chief Engineer of Test Equipment Ltd., Crawley.

E. H. Holder (*Associate Fellow*) has been appointed Manager of the Royal Ordnance Factory, Radway Green, Crewe.

D. James (*Associate Fellow*) has taken up an appointment as designer with the National Research Council of Canada.

R. W. Seymour-Lee (*Associate Fellow*) has recently joined Western Manufacturing (Reading) Ltd. as Chief Designer and Development Engineer.

Problems of Aircraft Production

Another Conference on "Problems of Aircraft Production" is being held by the Institution of Production Engineers at the University of Southampton on 18th and 19th December.

The following is the programme:

Friday 18th December 1953

Opening Luncheon: Principal Speaker, The Rt. Hon. A. T. Lennox-Boyd, P.C., M.P., Minister of Transport and Civil Aviation.

Session I: 2.45 - 4.30 p.m.
"The Trend of Design"
Speaker: D. Keith-Lucas, B.A., M.I.Mech.E., F.R.Ae.S., Chief Designer, Short Bros. & Harland Ltd.

Session II: 5.00 - 7.15 p.m.
"Production Problems Arising from the Trend of Design"
Speakers: G. H. Dowty, F.R.Ae.S., Chairman and Managing Director of Dowty Equipment Ltd. S. P. Woodley, M.B.E., Superintendent, Supermarine Division, Vickers-Armstrongs Ltd.

Saturday 19th December 1953

Session III: 9.15 - 11.15 a.m.
"User Problems Affecting the Industry"
Speakers: B. S. Shenstone, M.A.Sc., A.F.I.A.S., F.R.Ae.S., Chief Engineer, British European Airways. Air Marshal Sir Robert O. Jones, C.B., A.F.C., B.A., D.I.C., M.I.Mech.E., F.R.Ae.S., Air Ministry.

Summing Up: 11.45 - 12.45 p.m.
Sir Frederick Handley Page, C.B.E., F.C.G.I., Hon. F.I.A.S., Hon. F.R.Ae.S., Chairman and Managing Director, Handley Page Ltd.

The Conference fee is 25s., which includes the Luncheon to be held on Friday 18th December and refreshments during the course of the Conference.

Members of the Society are invited to attend and applications for tickets should be made to the Secretary, Institution of Production Engineers, 36 Portman Square, London, W.1. The closing date for applications is *Monday 7th December*.

A Lecture on the Wright Brothers

A lecture on "The Work of the Wright Brothers for Aviation" will be given by Capt. J. L. Pritchard, C.B.E., *Honorary Fellow*, to the Royal Society of Arts, on 16th December, at 2.30 p.m. Lord Sempill, *Fellow*, will be in the chair.

Applications for tickets should be made to the Secretary, Royal Society of Arts, 18 John Adam Street, Adelphi.

Leverhulme Research Fellowships 1954

These Fellowships and Grants are intended for senior workers who are prevented by routine duties or pressure of other work from carrying out research. They are limited to British-born subjects normally resident in the United Kingdom. In exceptional circumstances the Trustees may waive the condition as to residence.

Forms of application may be obtained from the Secretary, Miss M. Branney, Leverhulme Research Fellowships, 3/5 Salisbury Square, London, E.C.4 Telephone: City 1910. Applications must be received by *31st December* 1953.

Diary

LONDON

NOVEMBER 10th
SECTION LECTURE. Gusts and their Measurement. J. Taylor, M.A., A.F.R.Ae.S. In the Library, 4 Hamilton Place, W.1. 7 p.m.

November 12th
MAIN LECTURE. Stability and Control in Aircraft Design. J. C. Wimpenny, A.F.R.Ae.S. At the Institution of Mechanical Engineers, Storey's Gate, S.W.1. 6 p.m. (Tea at 5.30 p.m.)

November 19th
SECTION LECTURE. Aircraft Stability and Control. A. W. Babister, M.A., A.F.R.Ae.S. 4 Hamilton Place, W.1. 7 p.m.

November 26th
GRADUATES' AND STUDENTS' SECTION. The Case for the Light Fighter. P. A. Norman, Student R.Ae.S. 4 Hamilton Place, W.1. 7.30 p.m.

DECEMBER 1st
SECTION LECTURE. Integral Construction. K. L. C. Legg, A.F.R.Ae.S. 4 Hamilton Place, W.1. 7 p.m.

December 10th
MAIN LECTURE AT THE BELFAST BRANCH. Training Devices for Air Crew. C. B. Ringham and Dr. A. E. Cutler. Whitla Hall, Methodist College, BELFAST. 7 p.m.

December 15th
SECTION LECTURE. Jet-Driven High-Speed Wind Tunnels. D. L. Ellis, F.R.Ae.S. Hamilton Place, W.1. 7 p.m.

JANUARY 7th 1954
MAIN LECTURE. LECTURE TO YOUNG PEOPLE. The Fun of Finding Out in Flying. Sir Leonard Bairstow, C.B.E., F.R.S., Hon.F.R.Ae.S. At the Institution of Mechanical Engineers, Storey's Gate, S.W.1. 3 p.m.

BRANCHES

NOVEMBER 13th
Chester.—Annual Dance at Quaintways Restaurant, Chester.

November 14th
Brough.—Visit to the Chadderton Works of A. V. Roe & Co. Ltd., Manchester.

November 16th
Halton.—Branch Night. Branch Hut, R.A.F. Station, Halton. 6.45 p.m.

November 18th
Manchester.—Recent Development in Gliding. A. H. Yates. Reynolds Hall, College of Technology, Manchester. 7.30 p.m.

November 20th
Leicester.—Wind Tunnels and Wind Tunnel Testing. G. Beach. Loughborough College. 7.15 p.m.

November 23rd
Halton.—Junior Members' Night. Branch Hut, R.A.F. Station, Halton. 6.45 p.m.

November 24th
Boscombe Down.—Inaugural meeting of the Branch, with an address by Sir Frederick Handley Page, C.B.E. 5 p.m.
Henlow.—Films. R.A.F. Technical College, Henlow. 7.30 p.m.

November 25th
Bristol.—Operational Requirements in the R.A.F. Air Cdre. W. H. Kyle, C.B., C.B.E., D.S.O., D.F.C.. Conference Room, Filton House, Bristol Aeroplane Co. Ltd. 6 p.m.
Weybridge.—The Suppression of Jet Noise. Professor E. J. Richards. Vickers-Armstrongs. 6 p.m.

November 27th
Birmingham.—Some Uses for the Small Gas Turbine. Birmingham Chamber of Commerce. 7.30 p.m.

November 30th
Halton.—Films. Branch Hut, R.A.F. Station, Halton. 6.45 p.m.

DECEMBER 2nd
Brough.—From Subsonic to Supersonic. Dr. D. W. Holder. Electricity Buildings, Hull. 7.30 p.m.
Chester.—Rocket Propulsion and Interplanetary Flight. A. V. Cleaver. Grosvenor Hotel, Chester. 7.30 p.m.
Luton.—Short Haul and Charter Operations. George Hotel. 7.30 p.m.

December 4th
Bristol.—Engineering Associations' Joint Dance. The Royal Hotel, Bristol. 7.30 p.m.

December 7th
Halton and Henlow.—AREA MEETING. Some Thoughts on the Design of Fast Load-Carrying Aeroplanes. G. H. Lee, F.R.Ae.S. Branch Hut, R.A.F. Station, Halton. 6.45 p.m.

December 8th
Bristol.—Fly-Past, Present and Future. Captain J. Laurence Pritchard, C.B.E. Conference Room, Filton House, Bristol Aeroplane Co. Ltd. 6 p.m.
Henlow.—The Training of Test Pilots. Group Captain S. Wroath, A.F.C. R.A.F. Technical College, Henlow. 7.30 p.m.

December 9th
Manchester.—Principals of Weight Control. F. Howarth, A.F.R.Ae.S. Reynolds Hall, College of Technology, Manchester. 7.30 p.m.
Weybridge.—R. K. Pierson Memorial Lecture. Moral Outlooks in Aeronautics. Capt. J. L. Pritchard, C.B.E., Hon.F.R.Ae.S. Vickers-Armstrongs. 6 p.m.

December 10th
Belfast.—MAIN SOCIETY LECTURE. Training Devices for Air Crew. C. B. Ringham and Dr. A. E. Cutler. Whitla Hall, Methodist College, Belfast. 7 p.m.

December 11th
Preston.—Dinner Dance in Celebration of the Wright Brothers' Golden Jubilee. Imperial Hotel, Blackpool.

December 14th
Halton.—Junior Members' Night. Branch Hut, R.A.F. Station, Halton. 6.45 p.m.

December 17th
Bristol.—Celebration of the 50th Anniversary of Powered Flight.
Chester.—Dinner and Film Show to Celebrate the 50th Anniversary of Powered Flight. Blossoms Hotel, Chester. 7.30 p.m.

December 18th
Birmingham.—Film. 50 Years of Powered Flight. Birmingham Chamber of Commerce. 7.30 p.m.

December 21st
Brough.—50th Anniversary of Powered Flight Commemoration Dinner. Guildhall, Hull. 7.15 p.m.

JANUARY 4th 1954
Bristol.—A Review of Developments in Aircraft Equipment. G. H. Dowty. Conference Room, Filton House, Bristol Aeroplane Co. Ltd. 6 p.m.

January 6th
Brough.—New Developments in Aircraft Production Engineering. Professor J. V. Connolly. Electricity Buildings, Hull. 7.30 p.m.
Chester.—Guided Weapons. D. J. Lyons. Grosvenor Hotel, Chester. 7.30 p.m.
Luton.—Wilbur Wright Centenary Film. George Hotel. 7.30 p.m.

Elections

The following is a list of new members and transfers of membership of the Society:—

Associate Fellows

Walter Wallace Coles
Norman Rupert Meggs
Grevor Chilton Molyneux
Ivor Edward Moss (*from Graduate*)
Kenneth Robert Arthur O'Brien
Godfrey Nugent Wells

Associates

Arthur James Coghlan
Trevor Alfred Gallagher
Francis Ronald Minton
Frederick William Stock

Graduates

Reginald Mark Diamond (*from Student*)
Kenneth Ernest Yates

Students

Bernardus Hermanus Booyens
Christiaan Gabriel Du Toit
Robert Philip Francis
Ronald Bruce Glass
Dale Graham Hobson
Peter Exton Turner
Sarel Petrus Frederick Jansen Van Rensburg
Balakrishna Venkataraman

Annual Subscriptions

Members are reminded that their annual subscriptions become due on 1st January 1954. The rates are:—

	Home	Abroad
	£ s. d.	£ s. d.
Fellows	5 5 0	4 4 0
Associate Fellows	4 4 0	3 3 0
*Associates	3 3 0	3 3 0
Graduates (aged under 26)	2 2 0	2 2 0
Graduates (aged 26 and over)	2 12 6	2 12 6
Students (aged under 21)	1 1 0	1 1 0
Students (aged 21 and over)	1 11 6	1 11 6
Companions	3 3 0	3 3 0
Founder Members	2 2 0	2 2 0

* Any Associate elected before 1st October 1947 may, if he wishes, elect not to receive the Journal, and in this case his subscription will be reduced by £1 1s. 0d. to £2 2s. 0d.

It will avoid delay and confusion if members, when sending remittances for subscriptions, will state their names clearly and give their addresses and grades of membership. Remittances should be made payable to the Royal Aeronautical Society.

The Cierva Memorial Prize Essay Competition

In commemoration of the pioneering work of Senor Juan de la Cierva, and for the encouragement of the younger generation, the Helicopter Association offers annually two prizes for original essays of a technical nature on selected aspects of rotary wing flight. This competition is not restricted to Members of the Association, but is open to entrants of all nationalities, who must be under 25 years of age.

First and Second Prizes of £35 and £15 will be awarded to the competitors submitting the two best papers on either of the following subjects:—

"Turbines for Large Helicopters."

"Helicopter Performance Estimation Methods; a Critical Survey."

The closing date for entries is 31st December 1953. Applications for rules and entry forms should be made to the Honorary Secretary, The Helicopter Association, Londonderry House, 19 Park Lane, London, W.1.

Changes of Address

To assist in keeping the records of members correct and up to date the Secretary will be glad if all members will notify him as soon as possible of changes of address.

When notifying changes please give the following particulars:—

Name (in block letters). New address (in block letters).
Grade of membership. Old address.

Changes of address must be received *before the 15th of the month in order to be effective for the* Journal for the following month.

World Power Conference 1954

A Sectional Meeting of the World Power Conference will be held at Rio de Janiero, Brazil, from 25th July to 8th August 1954. Papers on eleven subjects will be presented by the British National Committee to the Conference. On "British Developments in Gas Turbines," an "Introduction" will be given by Sir Harold Roxbee Cox, *Fellow*, Chief Scientist, Ministry of Fuel and Power, and Professor W. R. Hawthorne, *Fellow*, Professor of Applied Thermodynamics, University of Cambridge, will give a paper on "Gas Turbines for Aviation."

Full particulars of the Conference may be obtained from The Secretary, British National Committee, World Power Conference, 201-2 Grand Buildings, Trafalgar Square, London, W.C.2.

Commonwealth Bursaries Scheme

The Royal Society and the Nuffield Foundation have jointly decided to initiate a Commonwealth Bursaries Scheme to provide facilities for increasing the efficiency of investigators by enabling them to pursue research, learn techniques or follow other forms of study. The scheme will operate for an experimental period of five years. Applicants must be sponsored by a recognised research authority and must produce evidence that he or she has prior permission to work in the laboratory or other scientific institution chosen.

Applicants will be considered, initially at six-monthly intervals, beginning early in 1954. Application forms and details of the scheme may be obtained from the Assistant Secretary, The Royal Society, Burlington House, Piccadilly, London, W.1.

Journal Binding

Self-Binder Cases

Self-Binder cases of the "Easibind" type are available from the offices of the Society. These binders are for members who do not have their Journals permanently bound, or who wish to keep their Journals together during the year for binding later.

These cases will hold 12 Journals which are kept in place by means of flexible steel wires. Journals can be inserted or withdrawn easily without damage, so preserving the contents for permanent binding later. The Journals will open flat at any page.

The binder is strongly made in durable dark blue leather cloth on stiff board covers and has gold lettering on the spine. The year is not blocked on the spine but there is a panel on which members who wish to use the binder as a permanent case can put the date.

The cost is 11s 6d. each including postage and packing for either the size to fit 1952 and previous Journals, or for the size to fit the Journal from January 1953, which has been increased in size. Orders and remittances should be sent direct to the Secretary at the Offices of the Society and it is *important to state whether the old size or new size is required.*

Permanent Binding

There is no increase in the price of permanent binding of Journals. The prices are:—

1952 Volume (including packing and postage) 16s. 0d.
Previous Volumes (including packing and postage) 18s. 0d.

Journals, *with a note of the name and address of the sender,* should be sent direct to The Lewes Press, Friars Walk, Lewes, Sussex, and the remittance to the Secretary at the Offices of the Society.

Acknowledgments

The Council acknowledge with thanks the return of back numbers of the Journal from J. D. Frier, Esq., *Fellow*; C. Gilford, Esq., *ex Associate Fellow*; and D. G. Hitchcock, Esq., *Associate Fellow*.

Journal of The Royal Aeronautical Society

WITH WHICH IS INCORPORATED THE INSTITUTION OF AERONAUTICAL ENGINEERS

VOLUME 57 NOVEMBER 1953 NUMBER 515

Bending Moments in Fuselage Frames Obtained by the Photoelastic Method

by

P. P. BENHAM, M.Sc., Grad.R.Ae.S.

(*Lecturer in Applied Mechanics, Imperial College*)

SUMMARY:—The experimental work described in this paper, which was divided into two sections, was done to determine the usefulness of photoelasticity as a method for analysing fuselage frame problems. In the first part, a series of tests was made by the photoelastic method on models without any supporting shell, which were in equilibrium under the applied loads, to determine the bending moment distribution in various types of frames and rings.

In all cases the theoretical bending moments were calculated from the principles of minimum strain energy. The experimental results agreed with theory to within 5 per cent. except close to the null points and joints.

The second part was devoted to shell-supported models, and the shear loads were reacted out through the skin to firm supports at the end of adjacent bays. Again the bending moments were determined for the entire frame from the isochromatic fringe pattern.

The shell-supported tests proved to be of considerable interest when compared with the theoretical curves of Ref. 4 for the corresponding relative stiffness parameters. In some cases agreement was good but in others there were marked differences. A possible reason for this was that compression effects in the skin were altering the distribution and value of the bending moments in the frame.

It was felt that there was plenty of scope for more interesting work on fuselage frame analysis, from the standpoint of shear flow determination, intermediate frame reaction, and skin-to-frame relative stiffness effects. All of these might feasibly be examined photoelastically with a certain degree of advantage over lengthy and laborious theoretical investigations.

1. Introduction

Mechanical testing is always considered to be an important part of engineering design with a more particular and special application in aeronautics because of the greater number of uncertain quantities. However, in the last decade or more the scope of the testing has been extended considerably from basic tests of ultimate strength and such like, to the practical establishment of experimental stress analysis. This has been due in the main to the increase in the number and complexity of the aircraft designers' problems, with the greater speeds and higher loads of present-day requirements.

Solutions to the problems from the theory of elasticity have often become lengthy and laborious, with consequent possibility of errors, and the need of approximations. This is quite apart from the increase in cost from the time involved for projects to pass through the technical offices.

Of the many methods of experimental stress analysis, the use of strain gauges for the determination of stress distributions seems to be the only one which is generally accepted and in wide use. Its main disadvantage appears to be that, in examining a number of large structural components, it is a costly and lengthy procedure.

The photoelastic method of stress analysis, which has not yet established itself in the industry, can be of practical value in some of these problems, as pointed out by Col. H. Jessop[1]. It has certain disadvantages and limitations, but it can be applied to a large variety of problems, and can provide accurate information of considerable value, quickly and inexpensively.

The problem tackled in this paper is that of the fuselage frame. The work was done[2] as part of the requirements for the degree of M.Sc., and so the results and experiments have had to be cut down in order to be presented in a paper of suitable size and scope; the original investigation itself was not as exhaustive as might be desired, because of the time limitation.

Since very little work has been done using the photoelastic method of stress analysis on aircraft structures, it was felt that the first steps were to investigate a technique of examination, the more general type of stress distribution to be expected in a frame, and the accuracy of the method. To this end the work was divided into two parts. Part I, as a preliminary study, concerned photoelastic model fuselage frames partially to scale, and loaded symmetrically so that a supporting shell was unnecessary. This part ends with a scale model test on a fuselage frame of a present-day production machine.

Having met with a fair degree of success, both in technique and results, a more ambitious approach was decided upon for Part II. In this, an attempt was made to compare model shell-supported frame conditions with those of a prototype and to determine stress distributions brought about by skin reactions. Again a scale model test was used in determining the conditions occurring in an existing frame prototype.

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Notation

p, q	principal stresses
σ	normal stress
τ	shear stress
σ_1, σ_2	inner and outer boundary stresses at any one frame section
n	fringe order
n_1, n_2	inner and outer boundary fringe orders corresponding to σ_1 and σ_2
F	model fringe value
f	material fringe value
M	bending moment
M_p, M_t	photoelastic and theoretical frame bending moments respectively
P	applied load on model
P_e	end load in frame
S	shear force in frame
I	moment of inertia of frame cross section
A	area of frame cross section
y	distance of any fibre from frame neutral axis
b, t	thickness of model and frame section
h	depth of frame section
R	radius of frame measured to neutral axis of section
t_s	supporting shell thickness
L	length of shell for one bay, *i.e.* between frame and end support bulkhead
θ'	isoclinic parameter. Direction of principal stresses measured to a vertical reference axis
ϕ, θ	angles defining points on the frame
θ_p, θ_t	position of photoelastic and theoretical null points respectively measured to a vertical reference axis
d_p	relative stiffness parameter $[=R^3tG/(EIL)]$
E	Young's modulus
G	shear modulus of rigidity
K', K''	constants, equal to $h^2f/12$ and $hf/2$ respectively

2. Previous Work

Although a full list of references has been given at the end of the paper, it may be of interest to summarise briefly what little photoelastic work has already been done on the fuselage frame.

In 1942, B. F. Ruffner[3] made a series of photoelastic investigations on shell-supported rings and bulkheads. The set-up was arranged as a cantilever with the bulkhead at the free end and the shell built in rigidly at the other. The main result obtained was the maximum bending moment in the ring. Comparing this with the corresponding value obtained from the curves in Ref. 4 for the correct relative stiffness parameter, shows quite a large discrepancy. This probably arises from the stiffness of the model when arranged as a cantilever, with a very short length of shell; although from a photoelastic point of view this arrangement is necessary to avoid excessive deflections hindering the light path.

In 1947 H. Becker[5] investigated the stress distribution in a wing spar bulkhead. A supporting shell was not used and the bulkhead was in equilibrium under the action of opposed couples at the spar root. The model was to true scale of the prototype and the results determined were the base and crown bending moments and end load, and the maximum shear stresses in the bulkhead. Comparison with the strain energy theory was favourable except for an error of 47 per cent. in the crown bending moment attributed to cement slippage effects.

The most recent contribution to bulkheads and frames was made in 1948 by W. A. P. Fisher[6]. Although coming under the heading of rings with applied couples at the ends of a diameter and primarily referring to jet engine mounting rings, it had some bearing on frame problems for the experimental technique if nothing else. Fisher concentrated on determining the null points and agreement between photoelastic results and theory, using a "deflection compatibility" method, is extremely good.

3. Experimental Technique

3.1. PHOTOELASTIC BENCH

The polariscope in the photoelastic laboratory was of the transmission type, using approximately parallel light through the model. It was 12 ft. long and had a clear 6 in. diameter field. This size proved invaluable, allowing reasonably large models to be accommodated. A smaller field than this would have made construction of accurate models and interpretation of fringe patterns considerably more difficult.

Both a white light source and a mercury vapour lamp filtered to give green monochromatic light were available. The images of fringe patterns and isoclinics were presented at the end of the bench and were recorded either with a half-plate camera or by sketching directly on to a vertical or horizontal screen.

The loading frame was arranged for adequate movement in the three principal planes, and was capable of applying and measuring hydraulic axial dead loads up to 350 lb. The use of a water tank for applying the loads proved a great advantage over normal dead weight systems, allowing a smooth and gradual application of load, which could be easily controlled remotely by valves, so that the growth of the fringe pattern could be studied at all times.

Photographs of the isochromatic fringe pattern were taken on rapid orthochromatic 0·250 Kodak half plates. These are green-sensitive plates and give a high contrast when used in conjunction with a contrast paper for prints. This contrast is essential in accurate determination of the position of fringes, when they are very fine and close together, as will be seen in some of the photographs.

Details of the equipment may be studied conveniently in various other papers, but it may be of interest that the bench used in this investigation was designed and built in the establishment for a total cost of under £100.

Figure 1 shows a general view of the polariscope and its various components.

3.2. MATERIALS AND MODELS

Choice of material was restricted by availability and cost, since some of the better photoelastic materials originate in the United States. The most satisfactory

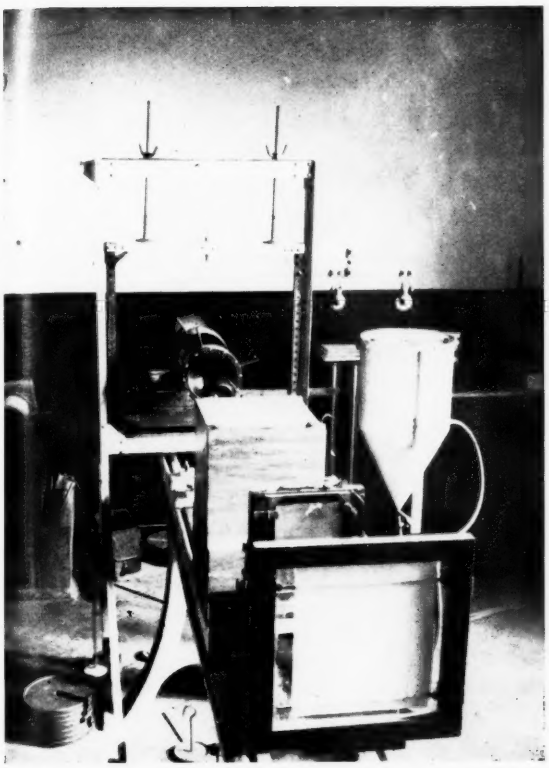

FIGURE 1. General view of the polariscope and its auxiliaries.

materials available in this country for photoelastic work are Marco, Catalin and Columbia Resin*. Of these three Marco is a casting resin which, although necessary for three-dimensional work, entails extra time and work unnecessary in two-dimensional applications, because the other materials may be obtained in readily prepared sheets of varying thickness and size. It was finally decided to use Columbia Resin C.R. 39.

In Part II the supporting shell was made from thin sheets of Xylonite having similar mechanical properties to that of C.R. 39.

The problem of making the models requires special note. The manipulation of a photoelastic material is in many ways very different from other types of model making and fitting. It was natural that a high degree of accuracy was required, but in addition great care had to be taken not to introduce machining stresses which could easily affect the true fringe pattern of stress on loading the model. C.R. 39 is an extremely brittle material and care had to be taken when machining to avoid chipping the edges. Another feature was the need to be prepared for a test within four to eight hours of the model's construction to avoid time edge effects.

Most of the models were turned out on a lathe or vertical miller, but for some of the more complicated frame boundaries, filing was done to avoid machining stresses or chipping when a model was nearly completed. An accuracy of $\pm 0{\cdot}001$ in. was attained when filing, and this was considered to be quite enough for the purpose.

*Since this paper was written Araldite has come into prominence as a photoelastic material.

4. Experimental Programme

In fuselage frame analysis the usual calculations made are to determine bending moment, shear force and end load in the frame section. Values of internal stresses are usually not of concern, unless there is some possibility of stress concentration.

Normal procedure in a photoelastic analysis is to obtain isochromatic fringe patterns of principal stress difference $(p-q)$ and the isoclinic map giving directions of principal stress. From these two diagrams or with the aid of an interferometer, the principal stresses have to be separated to give the required information. This operation can entail a fair degree of work and time, much of which is wasted if it is knowledge not really required by the designer.

Bearing these facts in mind it was decided to try to obtain the required results of bending moment and so on merely from a study of the fringe pattern and isoclinics without actually separating p and q. To do this, the main assumption entailed was that, because of the high ratio of radius of curvature to depth of section (of the order of 25:1 to 35:1 in large civil and military machines), "straight beam" theory could be adopted. If from experiment this proved to be the case, a great deal of time and labour would be saved.

A short elementary analysis will illustrate the method used. If σ_1 and σ_2 are corresponding inner and outer boundary stresses,

$$\sigma_1 = \frac{My}{I} + \frac{P_e}{A}$$

$$\sigma_2 = \frac{My}{I} - \frac{P_e}{A}$$

and hence $$M = \frac{I}{2y}(\sigma_1 + \sigma_2) = \frac{bh^2}{12}(\sigma_1 + \sigma_2).$$

σ_1 and σ_2 may be obtained directly from the isochromatic fringe pattern, being merely the model fringe value multiplied by the fringe order at the point,

i.e. $$\sigma_1 = Fn_1 \equiv \frac{fn_1}{t}$$

$$\sigma_2 = Fn_2 = \frac{fn_2}{t},$$

where f is the material fringe value and t the thickness of the model sheet.

Hence $$M = \frac{bh^2 f}{12t}(n_1 + n_2)$$

$$= \frac{h^2 f}{12}(n_1 + n_2) \text{ (since } b = t).$$

Since in the majority of the tests h and f are constant, the bending moment may be expressed as

$$M = K'(n_1 + n_2)$$

where $$K' = h^2 f / 12.$$

This expression for the bending moment is extremely convenient, since at any point on the frame it is only

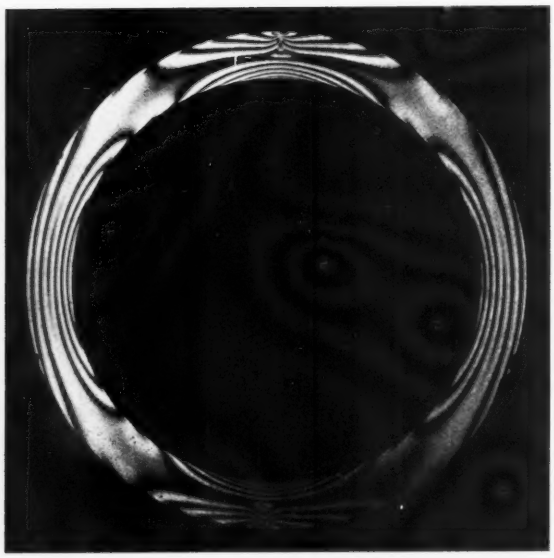

FIGURE 2. Fringe pattern of a slender ring compressed across a diameter. Test I.

necessary to read off the fringe order on the tension and compression boundaries from the isochromatic photograph.

4.1. TEST I

A preliminary test was done to determine the accuracy of this method. This used a slender ring compressed across a diameter. Although not a frame problem as such, the stress pattern is fundamental and appears in slightly different guises in the stress pattern of almost any shape of frame.

In addition to this the proportions of the ring were such as to represent the severest or smallest R/h ratio which would arise in the forthcoming experiments.

The fringe pattern and bending moment distribution for the ring are shown respectively in Figs. 2 and 3. A comparison of the bending moments, given in Table I, obtained photoelastically by the method already described, and the theoretical solution from principles of strain energy, show very close agreement, except for two sections close to the null points. However these may safely be neglected in comparison with the 2·77 per cent. and 1·52 per cent. errors in the two positions of maximum bending moment. As a further check, at one of the points of maximum moment, the stresses were separated using the standard slope equilibrium method (Fig. 4) and the moment was determined. There was found to be an error of 1·07 per cent., which is well within experimental limitations, and it appeared to justify the adoption of the simplified method of calculating bending moments.

4.2. TESTS II AND III

These tests, being similar, are coupled together and comprise the initial practical approach to a frame problem. They were intended to be partially representative

TABLE I

θ	n_1+n_2	M_p	M_t	*Percentage Error*
0	6·65	5·56	5·41	2·77
10	6·25	5·22	5·19	0·578
20	5·37	4·5	4·515	0·332
30	3·99	3·34	3·42	2·34
40	2·14	1·79	1·93	7·25
50	0·1	0·084	0·093	10·98
60	2·45	2·05	2·035	0·737
70	5·3	4.43	4·39	0·91
80	8·5	7·1	6·9	2·9
90	11·25	9·4	9·5	1·52

of a spar frame having opposed moments applied at the attachment points of the spar to the frame.

As there was no supporting shell there was no means of reacting the applied shear load; hence this aspect has been left until Part II. However, since frame bending moments are often calculated for the two conditions of applied load, shear and moment separately, these tests are of interest.

Test II was on a plain ring analogous to many civil machine frames of today. As expected the fringe pattern was almost identical to that of the ring in compression, only with signs of the stresses opposed in top and bottom halves. Stress peaks and hence maximum bending moments are readily seen, in Figs. 5 and 6, to occur on the vertical centre line and a line through the applied load points, although this latter moment has to be obtained by extrapolation because of the masking of the pattern by the loading arms.

Except for positions close to the null points, the percentage errors in Table II are very small on

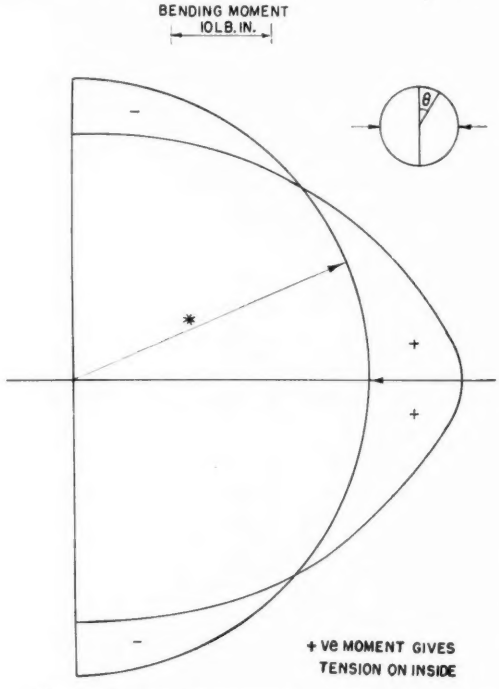

FIGURE 3. Bending moments for ring of Test I*.

* *Note:* The bending moment in this and other similar diagrams is measured radially from the centre corresponding to the particular arc.

TABLE II

θ	n_1+n_2	M_p	M_t	*Percentage Error*
0	—	—	17·5	—
15	10·96	11·85	11·74	0·935
20	9·15	9·9	9·91	0·101
30	5·92	6·41	6·4	0·156
40	2·85	3·09	3·225	4·18
50	0·342	0·37	0·488	24·2
60	1·57	1·7	1·73	1·73
70	3·14	3·4	3·37	0·89
80	4·15	4·49	4·37	2·74
90	4·35	4·7	4·705	0·106

comparing theoretical with experimental results. The average value for the location of the null points $\theta_p = 51{\cdot}84°$ is in close agreement with that obtained theoretically, $\theta_t = 51{\cdot}77°$.

Test III was also on a spar frame, but of a simplified contour of that which might be found in a large bomber fuselage. The frame model was made more flexible with a R/h ratio of very nearly 10:1 and a secondary minor arc was introduced to represent the bomb bay. Representation of the riveted joints used in practice was not attempted, although it would not have been impossible, and hence models were cut out complete. Couples were applied to the outer frame as in the previous test.

Stress peaks may be picked out from the fringe photograph (Fig. 7) and accuracy of results was again satisfactory but for one or two sections. The experimental bending moment distribution is shown in Fig. 8 and comparison with theory may be found in Table III. In this test, in addition to sections near the null points, those near the intersection of the bomb bay and outer

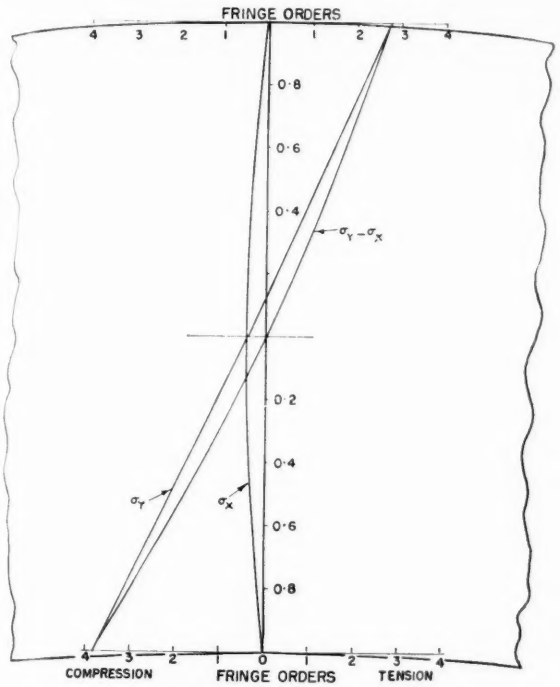

FIGURE 4. Separation of stresses at $\theta = 0°$ by the slope equilibrium method.

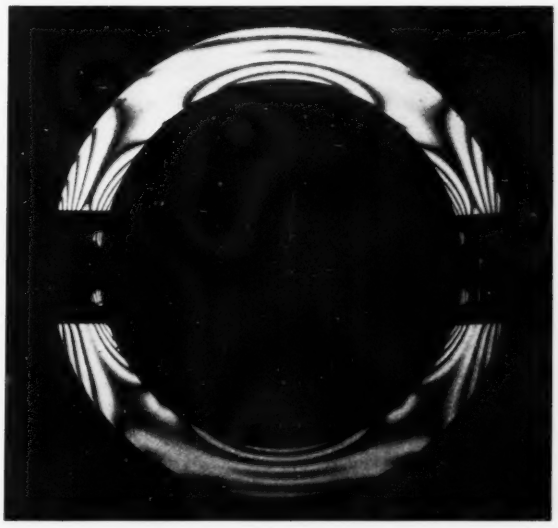

FIGURE 5. Fringe pattern of ring with opposed couples at the ends of a diameter. Test II.

frame digressed to some extent; this may be explained by the greater stiffness of the joints against the other parts of the frame.

4.3. TESTS IV AND V

The two final tests in Part I may be dealt with together. In Test IV the frame model examined was the

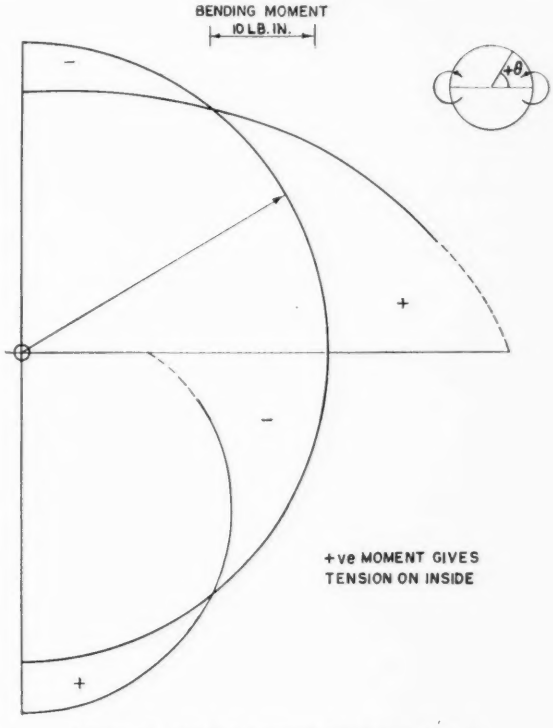

FIGURE 6. Bending moments for ring of Test II.

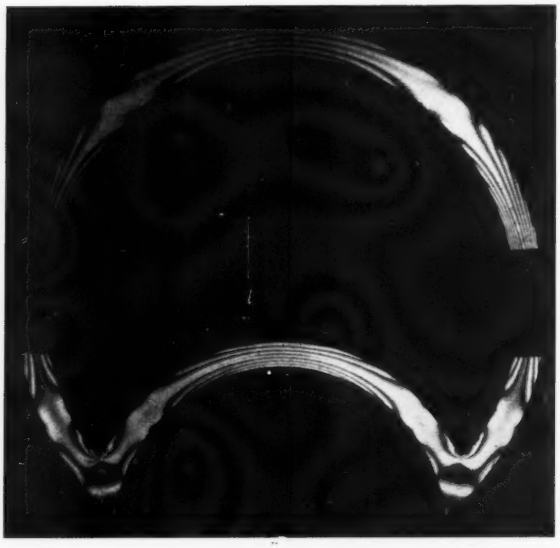

FIGURE 7. Fringe pattern of semi-representative spar frame with opposed couples at spar root.

last step before an attempt was made to determine the bending moments in a full size prototype frame.

It is appropriate to describe first the prototype and its loading condition, from which the derivation of the model for Test IV will then be seen.

When this investigation was decided on, a large aircraft firm suggested that it would be of great interest to examine practical cases, if that proved to be possible, and offered the necessary information on two bomber fuselage frames of a prototype of modern construction. The first, examined as Test V, was an intermediate frame and the second, dealt with in Part II, was a spar frame. It was felt that these two would give adequate and interesting scope for study.

The intermediate frame consisted of an outer frame, bomb bay side-wall and roof members, a central pillar and two tank support stays (Fig. 12).

The essential loading case was two outward reactions at the bomb door hinges caused by retraction and lowering of the bomb doors.

Bearing in mind the requirements of Test V, it was decided that perhaps a preliminary study on a more simplified scale would be an advantage. Model IV was designed partially to scale, but merely incorporated a circular outer frame joined to an arc representing the bomb bay side-wall and roof members, with a central pillar to give better moment distribution. By rotating the frame through 90°, the applied loads at the hinge points become a simple case of two opposing vertical forces in the straining frame.

If the central pillar had not been included, stress peaks would have occurred on the vertical centre line in both the outer frame and bomb bay, with the value of the bending moment building up far more rapidly in the bomb bay. The fringe photograph (Fig. 9) shows immediately, however, that the inclusion of the pillar served to shift the peaks in both the outer frame and bomb bay to some position on each side of the vertical centre line

TABLE III

OUTER FRAME

θ	n_1+n_2	M_p	M_t	*Percentage Error*
50	4·1	1·52	1·1	38
60	0·46	0·17	0·103	—
70	3·85	1·43	1·431	0·07
80	7·5	2·78	2·85	2·45
90	—	—	4·31	—
			6·44	
100	13·25	4·92	4·98	1·2
110	9·7	3·6	3·57	0·84
120	5·82	2·3	2·243	2·54
130	3·0	1·11	1·048	5·9
140	0·25	0·092	0·0043	—
150	2·1	0·78	0·83	6·02
160	3·7	1·373	1·458	5·8
170	4·85	1·8	1·833	1·8
180	5·3	1·965	1·961	0·24

BOMB BAY

θ	n_1+n_2	M_p	M_t	*Percentage Error*
110	3·8	1·4	1·242	12·7
120	1·5	0·556	0·322	—
130	0·82	0·304	0·505	39·6
140	3·15	1·17	1·232	5·02
150	4·85	1·8	1·812	0·662
160	6·1	2·26	2·24	0·892
170	6·75	2·5	2·49	0·401
180	7·0	2·6	2·585	0·58

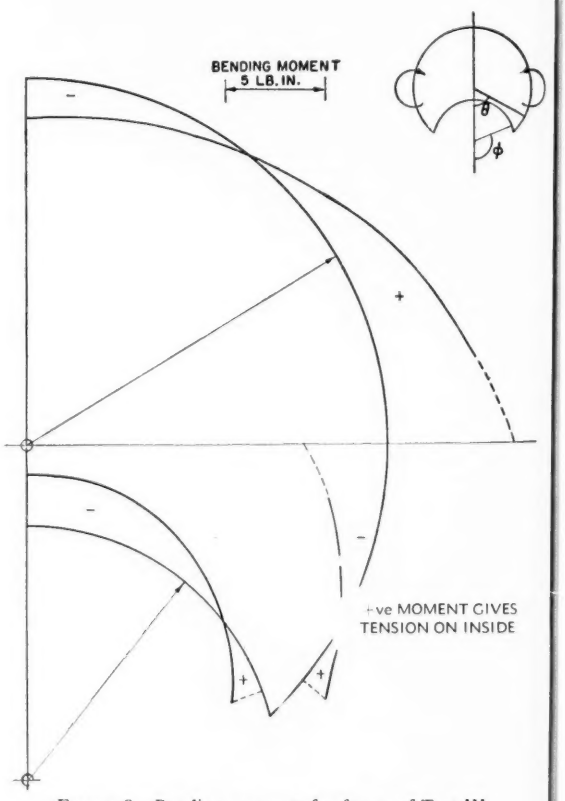

FIGURE 8. Bending moments for frame of Test III.

FIGURE 9. Stress pattern, for half fringes, of frame with outward applied loads at junction of the bomb bay and outer frame.

and the respective ratio of the order of the bending moment is approximately 2:3 (*see* Fig. 10 and Table IV).

The end load in the strut was determined directly from the fringe order, which being of a fractional value was ascertained using the Senarmont method of compensation giving a figure of 0·371 of a fringe.

This test also marked the turning point in the comparative time required to produce results by experiment and by theory. In the previous tests it would have been quicker to obtain results theoretically, rather than by experiment, and slightly more accurate, but in this test, by the introduction of another redundancy, *i.e.* the strut, the theoretical solution becomes more lengthy and laborious with a liability of error, yet the experimental aspect is no longer than before with the same accuracy. An interesting fact is the almost identical similarity between the left and right hand halves of the fringe pattern, which is as it should be. The internal regions of this model were filed to shape, the outer boundary being turned on a lathe.

The preliminary work having been completed, it was decided to proceed with the scale model test of the prototype frame previously described. The first problem was in making a model to scale; naturally it was impossible to reproduce a channel section with 16 gauge flanges to a scale of, say, 20:1. Secondly the size of the polariscope field made it impossible to scale down the prototype so that the whole model would be in the field, since the depth of section of the frame would have become approximately 0·1 in., which would have created difficulties in making the model and obtaining isoclinics.

Thus it became evident that a rectangular frame section would have to be accepted and a model made larger than the polariscope field, in order to obtain a workable depth of section. Hence the scale was arranged such that a quarter segment of the model could be investigated in the polariscope at one time. The model was then loaded up and the photographs taken of one segment; then, while still under load, the straining frame and the model were moved to bring the second

TABLE IV

OUTER FRAME

θ	n_1+n_2	M_p	M_t	*Percentage Error*
70	7·34	2·8	2·802	0·071
80	3·27	1·25	1·294	3·4
90	0	0	0·057	—
100	2·28	0·87	0·84	3·57
110	3·67	1·4	1·396	0·29
120	4·06	1·56	1·578	1·4
130	3·67	1·4	1·373	1·965
140	2·09	0·8	0·783	2·17
150	0·49	0·19	0·137	27·9
160	3·46	1·32	1·397	5·5
170	7·6	2·9	2·887	0·45

BOMB BAY

θ	n_1+n_2	M_p	M_t	*Percentage Error*
110	6·65	2·54	2·51	1·195
120	1·88	0·719	0·6	19·8
130	2·12	0·81	0·89	9·0
140	5·16	1·97	1·91	3·14
150	6·36	2·43	2·455	1·02
160	6·45	2·46	2·44	0·82
170	5·24	2·0	1·944	2·88

End Load in Centre Pillar

Photoelastic = 10·7 lb.
Theoretical = 10·48 lb.
Percentage Error = 2·1

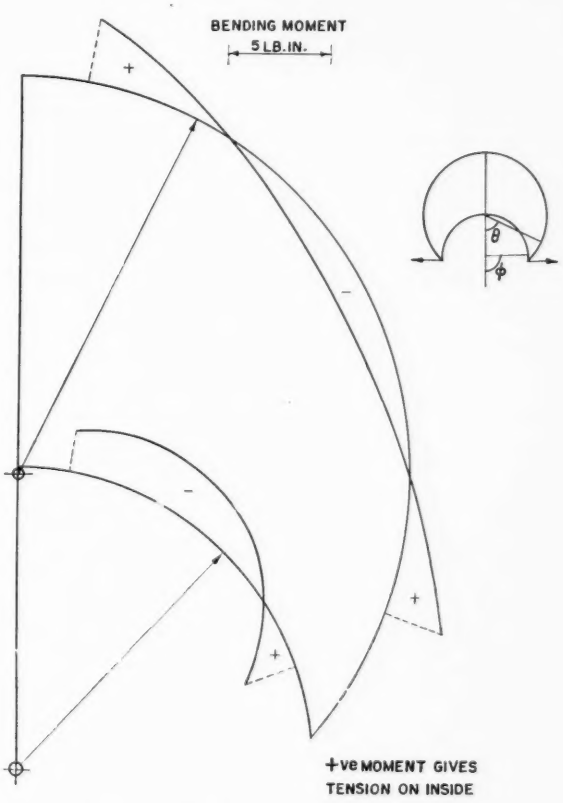

FIGURE 10. Bending moments for frame of Test IV.

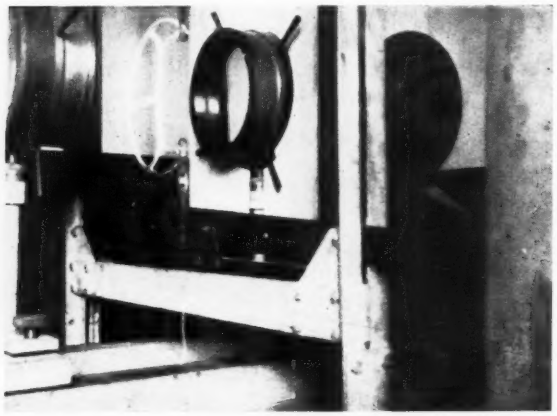

FIGURE 11. Scale model of prototype frame mounted in the straining frame ready for load application.

segment into the field. Since it was established that the pattern was symmetrical about a vertical centre line it was only necessary to protograph and sketch isoclinic maps of two segments. Fig. 11 shows the model in position in the straining frame ready for loading.

Another problem was that the prototype of extruded section had liners inserted at various points for strengthening purposes. Naturally it was desirous to represent this condition in some manner in the model, and so in addition to the linear scale, a scale of constant bending stiffness, *EI*, was introduced. Thus every section in the model was to the same scale of bending stiffness as the corresponding section in the prototype. All these problems may be simplified, if one convenient scale is chosen for the geometric centre lines of the frame so that the whole model may be inserted in the polariscope field. The depth of section is not kept to the same scale, but a suitable *EI* scale is chosen to give a workable depth of section, always bearing in mind that the latter must still be kept small in comparison with the mean radius of the frame.

In order to get a clearer picture of the stress pattern in the frame the photographs of the two segments have been joined together; unfortunately owing to slight aberration in the lenses at the edge, the sections do not mate up perfectly and the deflections apparent in the photograph, Fig. 12, do not appear in the loaded model.

The nature of the bending moment curve shown in Fig. 13 is rather as one would expect after the experience of the bending moment distribution obtained from the previous test. The only marked difference between the two was caused by the tank support stays which produced a change of sign in the bending moment in the outer frame on either side of the joint. The shape of the curve in the side-wall member is the same as in the bomb bay arc in Test IV. These similarities were encouraging and proved that the time spent on the previous experiment was not wasted.

Unfortunately details of the prototype, dimensions, load values or theoretical results cannot be revealed, but it may be said that the model bending moments were scaled up to prototype conditions and compared with the theoretical solution, which was made available. The comparison was very interesting and encouraging. Good agreement was obtained as to the location of the null points, or points of contraflexure, with the exception of one which, as may be seen, occurs in the outer frame near the top of the vertical pillar. This was not present at all in the theoretical curve. Some concern was felt as to the validity of this since the introduction of a redundancy, *i.e.* another strut, in those models concerned, always resulted in a change of sign in the previous bending moment in the outer frame with the accompanied points of contraflexure on either side of the joint. On the assumption that the model was incorrect due to the central pillar being too stiff, the only feasible query, the strut was gradually pared away in steps, the fringe pattern being noted each time, and only in the final extreme when the remains of the strut were completely cut away did that particular null point vanish. Thus representing a different condition altogether. As for the actual values of bending moment, these agreed reasonably in the roof member and the outer frame, except approaching the region of the upper

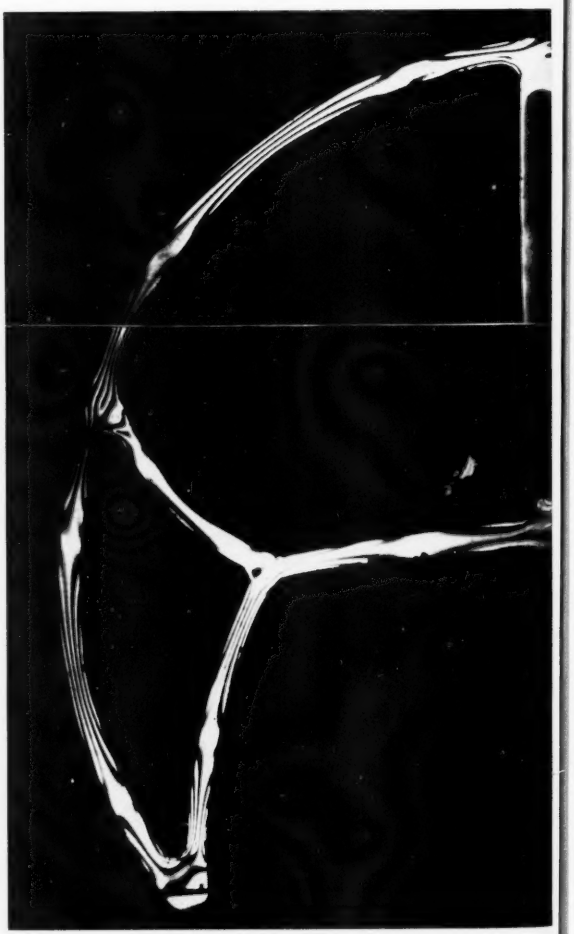

FIGURE 12. Stress distribution for scale model, Test V. Outward loads applied at bomb door hinges.

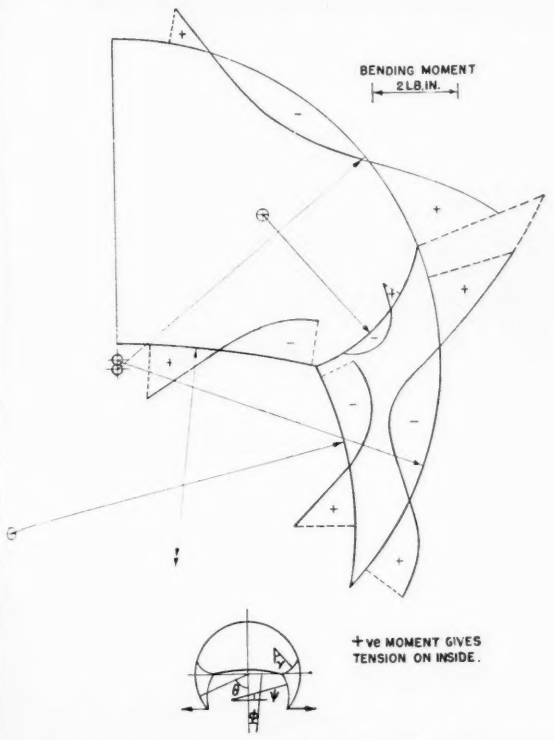

FIGURE 13. Model frame bending moments for Test V.

null point. The side-wall member was a little disappointing and this may have been caused by the greater stiffness of the joints at its upper and lower extremities.

4.4. SHEAR FORCE AND END LOAD

In all previous tests it was felt to be more satisfactory to compare theoretical and experimental results numerically rather than graphically, since at the majority of points the graphs would appear to be identical. The experimental curves of bending moment have been plotted to give some idea of the distribution in the various cases considered, which is sometimes difficult to see at first glance from the fringe pattern.

Shear force diagrams have not been included for the tests partly for reasons of space. However these may easily be determined from a combination of the fringe pattern and the isoclinics. The shear stress across any section may be found from the relationship

$$\tau = \tfrac{1}{2}(p-q)\sin 2\theta'$$

where τ is the shear stress, $(p-q)$ the principal stress difference and θ' the angle of inclination to some reference axis. These quantities are directly given photoelastically, $(p-q)$ from the fringe pattern, and θ' from the isoclinics. Having found the shear stress across any section it is then only necessary to determine the area under the curve to obtain the shear force.

End load may be determined by a similar analysis as for bending moment,

$$2\frac{P_e}{A} = \sigma_1 - \sigma_2$$

$$P_e = \frac{A}{2}(\sigma_1 - \sigma_2)$$

$$= \frac{Af}{2t}(n_1 - n_2)$$

$$= \frac{bhf}{2t}(n_1 - n_2)$$

$$= \frac{hf}{2}(n_1 - n_2) \text{ (since } b = t)$$

$$= K''(n_1 - n_2) \text{ where } K'' = \frac{hf}{2}.$$

Unfortunately this analysis does not give accurate results, as with bending moments, due most probably to the difference between similar values of fringe order $(n_1 - n_2)$. End load would best be obtained, when necessary, by integrating under the curve of direct stress at any section, but this is a longer process.

PART II

5. Experimental Procedure and Tests

In this part the more frequent type of frame condition is examined, when the applied loads are not in equilibrium and reacted by the frame itself, but the reaction is supplied as a shear flow in the surrounding skin and distributed along adjacent fuselage bays.

Until a few years ago, the normal assumption in problems of this kind was that the frame was rigid and that the shear flow distribution around the periphery was sinusoidal. However this was shown to be far from correct in most cases by Wignot, Combs and Ensrud and later by Goodey and others. It was proved mathematically that the majority of frames being far from rigid, the shear flow distribution could be vastly different from that assumed. The only experimental verification of this has been done briefly, using strain gauge methods, and was referred to in Ref. 7.

Furthermore these theories were primarily developed for completely circular flexible frames under various types of applied loading. In many other shapes of frames some approximation to the type of shear flow distribution is necessary. An incorrect assumption can produce large discrepancies in the resulting frame bending moments; it is hoped to show this in forthcoming tests.

Because of time limitations, the test programme in Part II could not be as exhaustive as might be desired. Thus it was decided to start with three tests on shell-supported rings under conditions of radial, tangential and moment applied loads respectively. Each model was to be designed and built to the same specification; it would then be possible to make a rapid check of the results with the theoretical curves given in Ref. 4.

5.1. MODEL CONSTRUCTION

The construction of the model frame has been discussed before but the problems involved and the method of construction of the shell unit are of practical interest.

FIGURE 14. Fringe pattern for shell-supported ring with outward applied radial load. Test I.

The material of the shell had to have closely similar elastic properties to that of C.R.39 from which the model frame was made, available in flexible sheets from 0·005 in. to 0·03 in. thick, and be easily and firmly attachable to C.R.39. C.R.39 could not be used because it is not produced in sheets less than $\frac{1}{16}$ in. thick and is rather costly if used for a purpose such as this. Finally Xylonite, which filled all the requirements listed, was chosen. The shell unit comprised two bays with the frame model situated in the middle; at the ends of the adjacent bays were bulkheads to carry away the input load and complete the unit. When the unit was ready for testing, the bulkhead at each end was bolted to a stand, firmly fixed to the bench. The stands were so arranged to permit travel of the polariser and analyser right up to the basic length of the shell unit.

A further question is that of obtaining free passage of the polarised light to pass through the inside of the shell and the frame model to its outer boundary, even when small deflections of the shell occur under load. This could only be overcome by making each bay of the shell slightly conical and joining at the centre line of the model. The taper on the shell was restricted to between 1° and 3°, it being felt that this small angle would be unlikely to disturb the general conditions of load and shear distribution. Skin butt joints were given uniformity where necessary with a 0·005 in. thick cover plate and all joints and fixations were made with Bexol acetate cement.

5.2. ANALYSIS FOR BENDING MOMENTS

In the method used for determining the bending moments in Part I there were no external stresses on the boundaries of the model except at the point of application of the loads, but in Part II, because of the shear flow reactions in the skin, the shear stress is not zero on the boundary of the model attached to the skin.

It has been shown elsewhere[3] that this shear stress is very small and for all practical purposes the normal stress on the boundary is equal to $(p-q)$, the principal stress difference. Hence $\sigma = F_n$ and the bending moments may be determined as in Part I.

5.3. TESTS I, Ia, II, AND III

These four tests were of the same nature, other than the loading conditions, and are discussed under the one heading.

The first consideration was the derivation of a representative or scale model. In Part I it was straightforward to produce a scale model frame, but now it is also necessary to represent skin and stringers by a thin sheet of Xylonite.

Naturally it would be unsatisfactory to distribute the cross-sectional area of the stringers round the periphery to give a new greater effective skin thickness, and then to use a linear scale to give a model skin thickness and bay length. Thus, since the final comparison was to be made with the work in Ref. 4 it was decided to adopt the criterion used there. This was a parameter "d_p" which expressed the shear rigidity of the shell in relation to the bending stiffness of the frame. This parameter, known as the "relative stiffness parameter," is given by

$$d_p = \frac{R^4 tG}{EIL}.$$

The model shell was then designed to have the same value of d_p as the prototype, the frame being made to scale as in Part I.

In these four tests then, the shell units, being designed similarly, had a value $d_p = 165$. This figure was not chosen in particular, but the variables R, t, I and L were arranged so that they should be within the scope of the apparatus and the materials available yet would provide a value of d_p between 100 and 200, representing a fairly flexible frame. This does not imply that a model could not be designed to cater for the parameter of a full size prototype but that it was safer to start the investigation with reasonably desirable conditions.

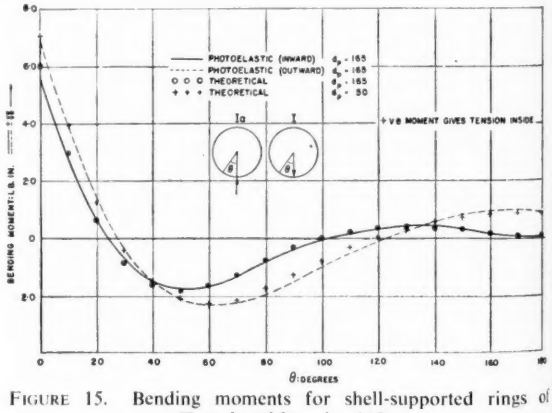

FIGURE 15. Bending moments for shell-supported rings of Tests I and Ia. $d_p = 165$.

n Test I the ring was loaded radially the direction of application being outwards from the centre. Calculating the bending moments from the fringe photograph of Fig. 14, and plotting them against the theoretical curve showed a marked difference (Fig. 15). In fact the experimental curve lined up almost exactly with the theoretical curve for a parameter of $d_p = 50$ rather than for $d_p = 165$, as was hoped and expected. This proved rather disappointing, and led to doubts as to the requirements for the complete model representation. Having gone this far it was decided to continue with Tests II and III and to see if the results were equally poor.

Test II considered the case of a tangential load applied to the ring. Stress peaks appeared at regions on either side and fairly close to the load point. In the half of the ring remote from the load point the stresses reduced rapidly to zero at the point diametrically opposite the load point. The fringe pattern was not quite symmetrical as it should be and a reason for this is suggested later.

Since the fringe pattern (Fig. 16) was not symmetrical about the centre line, bending moments were calculated and plotted for each half of the ring. In Fig. 17, the full line is the curve for the half (*a*) which is in the direction of the load, *i.e.* if the load acts tangentially from right to left with reference to a vertical centre line, the full line curve is for the half on the left and the chain dotted curve for the half (*b*) of the ring on the right.

It may be seen that the bending moments for half ring (*a*) agree quite well with the theoretical curve from $\theta = 0^\circ$ to $\theta = 90^\circ$, but differ somewhat between $\theta = 100^\circ$ and $\theta = 140^\circ$, the experimental values being higher over this region with a maximum difference of 30 per cent. The bending moments for the half ring (*b*) have altogether a different distribution and much lower values. It was found that this curve corresponded quite closely to the theoretical curve for a stiffness parameter $d_p = 50$ and an applied load of half the value used in the experiment.

The next step was to consider the case of pure moment applied to the ring forming Test III. The main difficulty found here was in the application of a pure moment only, without inadvertently introducing any radial or tangential component. Having devised a suitable rig for loading, a further problem arose from the resulting fringe pattern. An applied pure couple produces such a local form of stress pattern that fringes built up at an alarming rate close to the applied couple

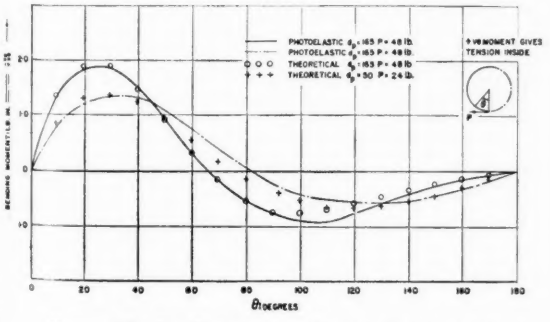

FIGURE 17. Bending moments for Test II. $d_p = 165$.

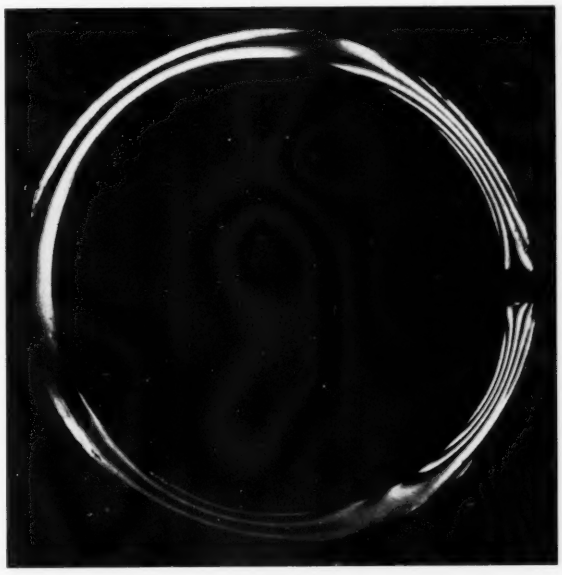

FIGURE 16. Stress distribution in shell-supported ring with applied tangential load.

with hardly any fringes appearing on the opposite side of the ring. Therefore it became necessary to use a compensation method in these regions of low stress.

The stress pattern (Fig. 18) was almost exactly symmetrical in each half and the resulting bending moment curve (Fig. 19) gave excellent agreement with that obtained theoretically for $d_p = 165$, except near the point of application of the load. This is because a theoretically applied moment acts at a point, whereas the experimental conditions require a finite arm in order to apply a moment.

Before attempting to interpret the results of these three tests, it was decided to check on the asymmetry of the stress pattern in the tangentially loaded ring, by arranging the load to act in the direction opposite to that used in Test II. It was found that the stress pattern was still slightly asymmetrical, but the stress distribution which had previously been located in the half ring (*b*), had moved to half (*a*) and vice versa. This confirmed that the construction of the model was not at fault.

By a similar reasoning, although theoretically the direction of the radial load in Test I, *i.e.* inward or outward, is immaterial as far as the value and distribution of bending moments is concerned, merely altering the sign, it was now felt necessary to check this experimentally. This was done under the heading of Test Ia. A quite surprising difference was found to exist in the stress distribution. It may be seen by comparing fringe photographs of I and Ia (Figs. 14 and 20) that the most immediate observed difference occurs in the second pair of null points remote from the load point. In I these were located at approximately $\theta = 120^\circ$ whereas in Ia they dropped to $\theta = 100^\circ$. In addition to this the stresses in the half of the ring remote from the load were considerably lower in value in Ia than in I. On plotting the experimental bending moments for the inward radial

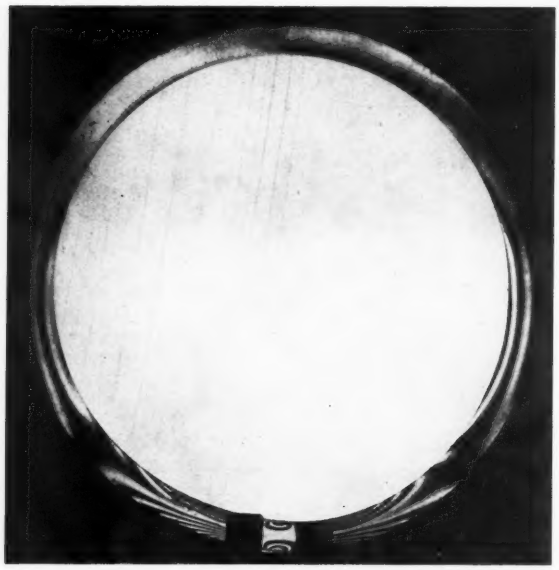

FIGURE 18. Fringe pattern for shell-supported ring with applied moment load.

load, there was found to be an excellent agreement with the theoretical curve for $d_p = 165$. It should be pointed out here that the fringe photograph of I was for an applied load of 20 lb. and that of Ia for a load of 25 lb. It was necessary to have a higher load in Ia to obtain the low value of fringes in the region $\theta = 140°$ to $\theta = 180°$. Therefore when plotting the bending moments the values of I were scaled up to the equivalent applied load of 25 lb.

It is also of interest to note here that in Ruffner's investigation, for the one shell-supported ring for which a complete bending moment curve is given, the relative stiffness parameter for the shell unit is $d_p = 30$ whereas the experimental curve agrees closely with the theoretical curve for $d_p = 0$, *i.e.* a rigid ring. All the tests were done for an outward radial load. Thus Ruffner obtained the same effect as occurred in Test I of the present investigation.

The initial tests in this part of the paper are very interesting and give rise to a certain degree of speculation as to the reasons for the variations in the results described; but it is felt that insufficient experimental work was done to justify any conclusive reasoning.

The following tentative suggestion is put forward that, when a shell-supported ring is loaded symmetrically, and theoretically a symmetrical stress pattern and bending moment distribution should result, in fact this is not so because part of the supporting shell is in a state of compression and is not transmitting its full share of the input load, while the remainder of the shell is in tension and is having a full load-carrying effect.

This would seem to be borne out partially by the fact that in Test III the applied pure couple imparted very little overall deflection to the shell, due to the nature of the load, and the resulting bending moments were quite accurate. Yet in Test I, Ia and II, deflections

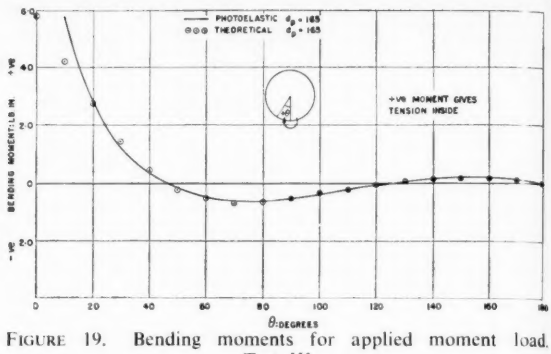

FIGURE 19. Bending moments for applied moment load. Test III.

of the shell due to the application of load were quite apparent. Some of the skin was in compression and part in tension, and the bending moments determined have a varying degree of agreement with theory. The model shell unit itself seemed satisfactory from this preliminary study, but it is quite possible that other designs and approaches to the problem might give more satisfactory results.

As it was necessary to prepare for the scale model investigation of a prototype, it was not possible to devote further time to this part of the investigation.

5.4. SHELL IN TORSION

To those who might be concerned with the problem of stresses induced into a frame by torsion in the supporting shell it may be of interest to mention a subsidiary experiment. This was not one of the main tests and was not examined in great detail.

A shell unit was constructed having a relative stiffness parameter of $d_p = 350$. Opposed couples were

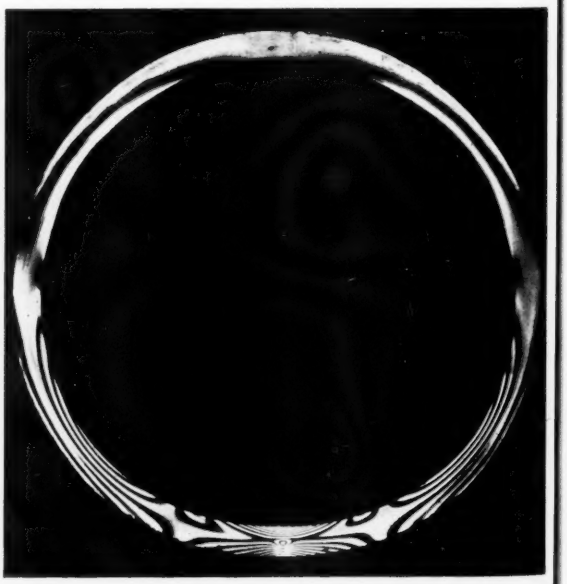

FIGURE 20. Stress pattern for shell-supported ring with inward applied radial load. Test Ia. $d_p = 165$.

applied to the bulkhead at each end of the shell so that the skin was in torsion. The torque was increased until buckling of the skin was just about to occur and at this point there was found to be quite a distinct stress pattern in the frame ring. The bending moment distribution gave a sinusoidal curve with four peaks of moment of equal value, and four null points, all the eight points being symmetrically situated round the ring at approximately 45° intervals.

5.6. TESTS IV AND V

These two tests were mainly to give a little more experimental information before embarking on the scale model prototype test.

The first test used a circular frame with a large cut-out, for a bomb bay, which was also circular. A more even distribution of load was obtained by having a vertical strut connecting the top centre of the bomb bay member to the outer frame. A radial load was then applied outwards from the centre of the bomb bay. For practical reasons, so that the stress pattern was not masked in the bay member, the load was applied through a pin located near the bottom of the strut (Fig. 21). The stress pattern is found to be somewhat like that in Test IV of Part I, which represented $d_p=0$ or skin thickness zero, while the model under consideration is considerably more flexible and is shell-supported. Although this made the model a little awkward to construct, the skin was attached to the bomb bay as well as the outer frame to preserve continuity and to obtain complete skin shear distribution. The shell unit is shown in Fig. 22, bolted in position to the stands in the straining frame before loading.

Test V was for a circular frame with a propped floor beam. Although theoretically a fairly straightforward problem, considerably more work is entailed than in, say, a simple ring with a radially applied load. The photoelastic approach, however, is just as simple and involves almost the same length of time in each case. The only extra work is in the construction for the floor and props. These were not an integral part of the frame but were prefabricated and cemented into position separately. Four props were used and a distributed load of 60 lb. was applied to the beam.

The fringe photograph (Fig. 23) shows maximum stresses ocurring a little above the floor beam, at the top centre of the frame, and at the junctions of the floor and props with the frame. This stress pattern gives rise to the general type of bending moment distribution to be expected.

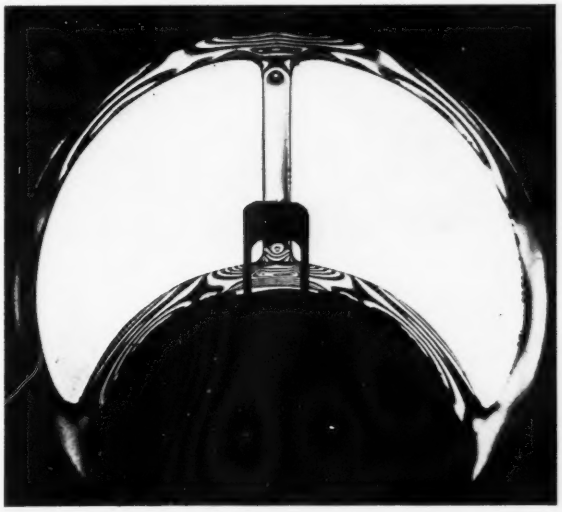

FIGURE 21. Photograph of half fringes for semi-scale bomber frame with outward applied radial load at base of central pillar. Test IV.

Theoretical bending moments as obtained from the curves of Ref. 4 are plotted with experimental curves for comparison for both Tests IV and V in Figs. 24 and 25.

5.7. TEST VI

To complete the experimental work it was felt that the most useful and practical step would be in the attempted determination of bending moments set up in an actual prototype. The case considered was that of one of the spar frames from the same machine as the frame chosen for Test V of Part I. The loading conditions were vertical shears at each root of the spar.

The design of the shell unit was based on the relative stiffness parameter as used in previous tests, as probably

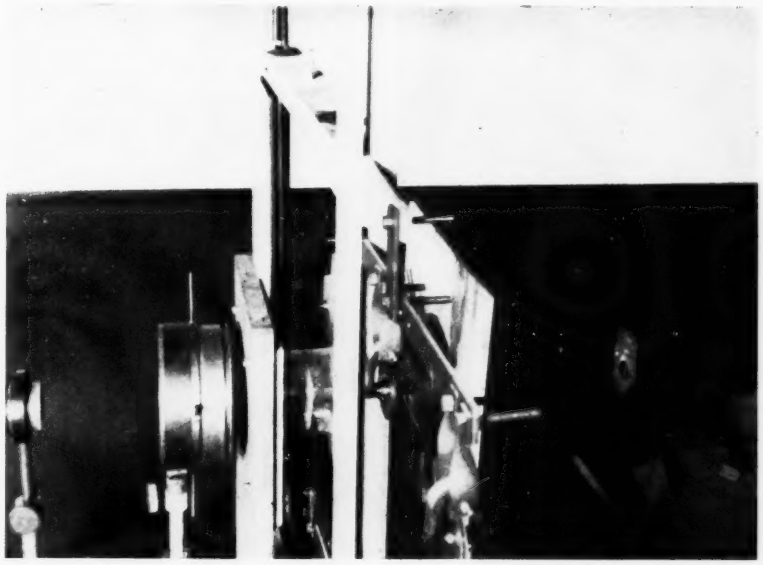

FIGURE 22. Model shell unit of Test IV bolted in position on the photoelastic bench before loading.

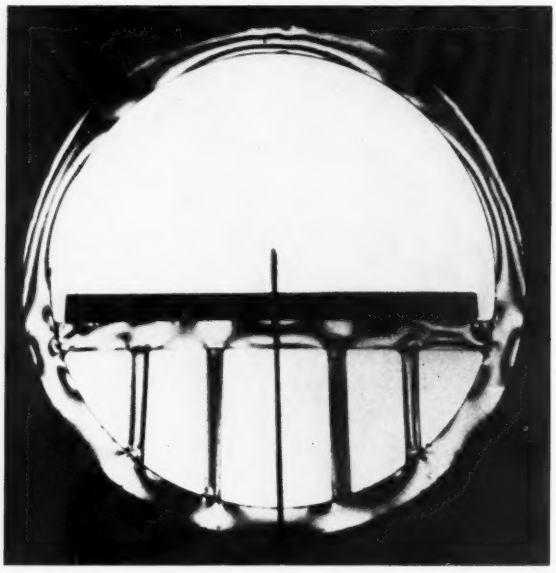

FIGURE 23. Stress distribution for half fringes in model frame with propped floor beam. Test V.

being the most reasonable way of representing full-scale conditions. To avoid moving the shell unit while under load, a rather awkward task, as in Test V Part I, the model frame was made such that it could be accommodated completely in the field of the polariscope (Fig. 26). A linear scale was used for the centre line of the frame and a scale of bending stiffness was used for the actual frame section, its depth being kept small in comparison with the radius. The spar was arranged to have the same stiffness relative to the frame section as in the prototype.

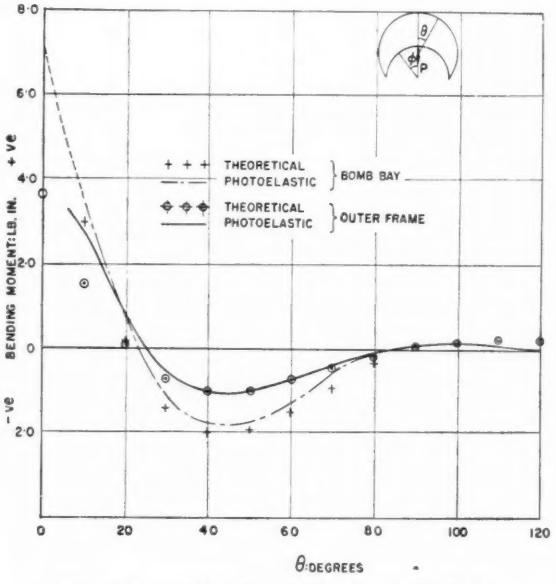

FIGURE 24. Bending moments for frame of Test IV.

It has not been permissible to plot or give values of the theoretical bending moments, but the experimental stress pattern and bending moment curves are shown (Figs. 27 and 28) for the model before applying the bending moment scale factor.

Close agreement was obtained for the portions of the curve round the outer frame both above and below the spar. Discrepancies occurred over the bomb bay side-wall and roof member, mainly because in the theoretical calculation the three ties between the spar and the bomb bay roof member were taken as pinned, while in the model they were fixed. From the fringe photograph it is seen that the left and right hand ties have taken considerable stress, thus altering the distribution in the side-wall and roof member.

After all photographs, readings and sketches had been taken, the applied loads were increased until failure occurred. This took place at one of the two points of maximum stress in the frame, but the shell did not fracture in itself and only detached from the frame in the region of the break. The stress distribution adjusted itself to accommodate the new conditions existing; this is well illustrated in the fringe photograph of Fig. 29.

6. Conclusion

The main purpose of this series of investigations was a probe into the usefulness of photoelasticity when applied to fuselage frame problems. A fairly wide variety of problems and conditions had to be examined and it was unfortunate that there was not enough time to go into some of the more interesting results in more detail, and to produce, say, families of curves for a large range of varying frame flexibility, as is frequently presented in theoretical analyses of this problem. It is usually during and after an investigation of this kind that many interesting possibilities for future examination and development arise, some of which are given later.

A summary of the experimental work shows that, in Part I excellent agreement is obtained between photoelastically determined bending moments and those found theoretically from principles of minimum strain energy. These are the cases which sometimes arise when the frame has applied loads which are in equilibrium and reaction takes place in the frame without an accompanying shear flow in the skin. Photoelastic models are

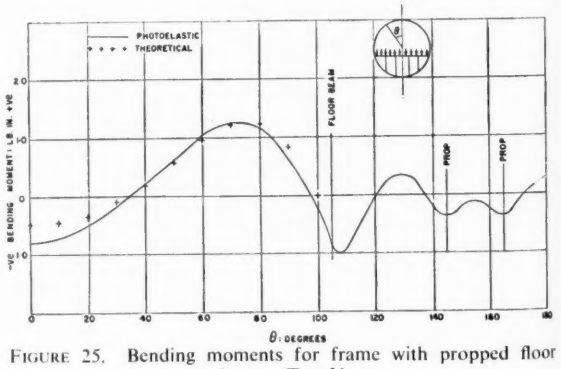

FIGURE 25. Bending moments for frame with propped floor beam. Test V.

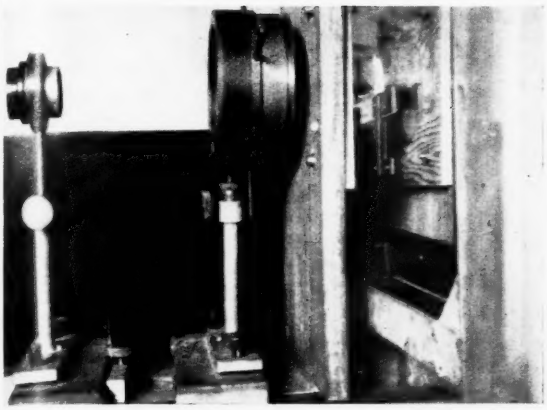

FIGURE 26. Scale model shell unit of bomber spar frame set up in straining frame ready for testing.

easily made and tested and results may be obtained quite rapidly.

The investigations of Part II are representative of the more general type of frame problem where shear has to be reacted by skin and stringers along the fuselage. The results of all the tests suggested that the shell unit as previously described might prove to be the most satisfactory yet simple way of representing prototype conditions, using the relative stiffness parameter d_p as the basic criterion for model design and similarity. Experimental bending moments in the frames proved rather erratic, some agreeing quite well with theoretical curves, others diverging somewhat. The cause could not be laid down with any degree of certainty but the trend of the results seemed to indicate that compression effects in the skin were having a marked influence on the stress distribution.

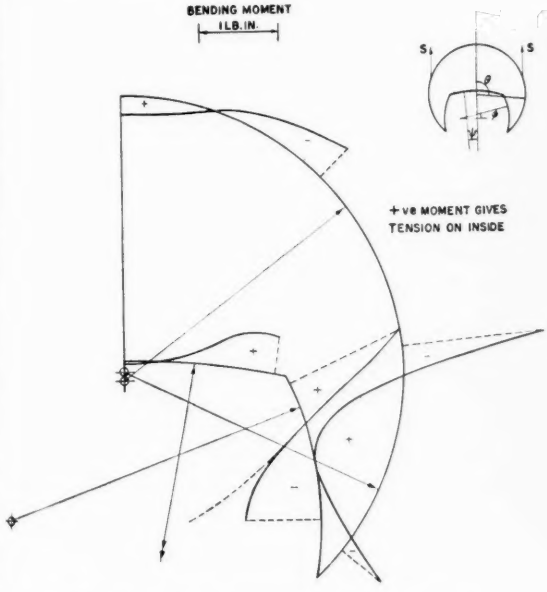

FIGURE 28. Bending moment for frame of Test VI.

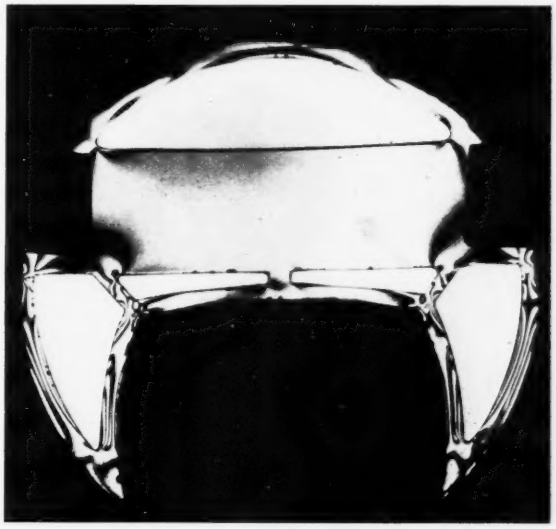

FIGURE 27. Fringe pattern of scale model bomber spar frame. Applied shears at spar roots.

It would be very interesting in the circumstances to note the effect of using stiffeners attached to the skin and frame, to represent stringers, which would help to transmit end load and to relieve compression in the skin.

Again in this part of the work tests could be carried out and results determined as quickly as in the simpler tests of Part I.

It is unfortunate that it was not possible to determine the shear flow round the frame by the experimental procedure adopted in this work, as this would have been quite a valuable result. However, a method of approach is suggested in the next section which might lead to its determination.

FIGURE 29. Stress distribution in scale model bomber frame after failure occurred in side-wall member.

Shear and end loads in the frame may be determined by the normal methods of photoelastic analysis, though the solution involves a little more work than that for bending moments. But since the photoelastic method is by no means intended to replace theory, an evaluation of the bending moments alone would be of sufficient use for comparison.

The photoelastic method of analysing frame problems has distinct possibilities and is a useful tool for the theoretical stressman. In simple cases it may be used as a rapid check. For complicated problems requiring a lengthy theoretical solution, with possible assumptions as to shear flow involving errors, a preliminary investigation could be made photoelastically to determine the type of bending moment distribution to be expected and hence by deduction the form of shear flow. The length of time for a completed test would be no more than for a simple case.

More experimental work similar to that of Part II is needed before quantitative results of bending moment for a prototype, obtained from a photoelastic scale model, could be accepted without question.

7. Suggestions for Further Study

(1) Determining the bending moment curves for radial, tangential, and moment loads for a range of stiffness parameters, $d_p = 0$ to 4,000.

(2) The effect of stiffeners to represent stringers attached to the skin, and frame, to see whether this overcomes the discrepancies which are possibly caused by skin compression effects.

(3) The stress distribution and moment in complete and cut bulkheads (as opposed to frames with shallow section), with and without cabin pressurisation.

(4) The stresses set up in frames adjacent to and away from a loaded frame due to the loads in the frame. These may be investigated experimentally by increasing the length of the shell unit to accommodate three or four frames. Then only the frame required to be analysed is made of a photoelastic material, the loaded frame and others being made of an optically insensitive material such as Perspex or celluloid.

These four problems may be investigated with the normal polariscope and general model arrangements described in this paper.

(5) Shear flow distribution in the skin. In this case it is required to analyse the stress pattern in the skin and polarised light must therefore be passed radially into the shell instead of axially through the shell, as for the frame.

This immediately gives rise to certain experimental difficulties, the most serious being the relative retardation produced in the skin under load. Naturally to produce any degree of model to prototype similarity the model shell must be very thin, of the order of 0·005 in. to 0·02 in. thick. If then, a material is used for the skin having, say, a fringe value of 50 lb./ in. per fringe, the model fringe value would be $f/t = 50/0{\cdot}005 = 10{,}000$ lb./in.^{2} per fringe. As the limit of fringe stress proportionality is of the order of 3,000 lb./in.^{2} a maximum of only 1/3 of a fringe could be obtained at best, thus requiring continuous use of a compensation technique. The only other solution is the use of a highly optically-sensitive material. At present the only photoelastic material available which might serve the purpose would be gelatine. Its fringe value and mechanical properties would depend on the mix, but for a mix having a fringe value of 0·12 lb/in.^{2} per fringe shear it would only be necessary to have a stress of 24 lb./in.^{2} in the skin to produce one fringe, which is a much more satisfactory arrangement. Naturally a correction would have to be made for the fact that Young's modulus for gelatine would be much lower than that for the photoelastic material used for the model frame.

There are two further major experimental points to be considered. Firstly, as the shell has curvature the path would have to be restricted to a narrow strip. Some arrangement could then be devised so that the shell unit revolved about its axis under load, and a cine camera would then record successive strips of shell pattern.

The second point is that when the light has passed from the polariser through the skin, it would have to be reflected either back radially or along the axis to the analyser, to avoid the light passing through the skin on the side of the shell opposite to its point of entry.

The five suggestions given for further investigation constitute a few interesting approaches to the frame problem by the photoelastic method. Yet this is but a part of the aircraft structure as a whole and there is no reason to suppose that photoelasticity could not have useful applications in many parts of the structure.

ACKNOWLEDGMENTS

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Types of Flow on Swept Wings

With Special Reference to Free Boundaries and Vortex Sheets

by

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SUMMARY:—This is a discussion of the various types of flow which may occur on thin swept wings when the air stream is not attached to the wing surface everywhere. In particular, the behaviour of vortex sheets which arise from such mixed flows is examined.

1. Introduction

In the past, the interest in the flow about wings as lifting surfaces has been limited largely to that type of flow in which the air stream is attached to the wing surface everywhere. This was sufficient for most practical purposes in the case of unswept wings of comparatively large thickness, and nearly all flight conditions were governed by that type of flow. Aerofoil theory as it is known generally assumes such a flow.

With the introduction of swept and thinner wings, other types of flow frequently occur and cannot be excluded from consideration. For example, a localised separation of the flow from the surface at the wing tip can no longer be ignored because its effect on the pitching moment can be appreciable when the wing is swept. Again, dead-air regions may be formed along the leading edges of thin wings as the result of a separation of the laminar boundary layer; these dead-air regions are separated from the main flow by a free boundary. The boundary layer at the rear of swept wings shows a sublayer in which there is a predominantly spanwise out-flow, and a separation of the turbulent boundary layer from the rear of the section may again lead to the formation of a dead-air region extending in this case into the wake. Matters are complicated by the fact that different types of flow may occur simultaneously on the same wing in which case there must be part-span vortex sheets separating the various regions from one another. Such vortex sheets normally leave a trace in the wake behind the wing additional to the ordinary sheet of trailing vortices. The occurrence of such mixed flows may have profound effects on the forces and moments on the wing as well as on the downwash at the tailplane. In reviewing the various possible flow patterns, it will be seen how precarious the existence of the common type of wholly attached flow really is, especially for thin or swept wings.

The present paper follows on the classical investigation into the stalling problem by Melvill Jones[1] 1934. Some of the possible types of flow are briefly considered together with the various vortex patterns which must arise if different types of flow exist on the same wing at the same time. The conditions which bring about such a mixed flow are not discussed. Too little is as yet known about them and further investigations are needed before such a discussion can take place. In the present note, however, it was thought worth while to remind ourselves of some of the accepted laws of vortex motions in general, in order to assist the understanding and the interpretation of the results from experimental investigations into the various flow patterns, *e.g.* flow visualisation.

In discussing such types of flow, simplifications and abstractions cannot easily be avoided, such as the concept of a thin vortex sheet representing a discontinuous change of the velocity. Discontinuities like this do not arise in a real flow; they are levelled down by turbulent mixing. However, such abstractions have proved very useful for theoretical analysis; the ordinary aerofoil theory provides a good example: the assumption of a thin sheet of trailing vortices in the direction of the free stream has proved a very useful abstraction in spite of its actual rolling up and dissolution by turbulent motion.

Notation

x, y, z	rectangular co-ordinates; $x=0$ at leading edge; x positive in the direction of the main stream
c	wing chord
b	wing span
t	wing thickness
A	aspect ratio
α	geometric angle of incidence
φ	angle of sweep of mid-chord line
V	velocity
v	velocity increment
V_0	velocity of the main stream
M	Mach number
γ	strength of vortex distribution
C_p	pressure coefficient
C_l	lift coefficient of section
C_L	overall lift coefficient of wing
C_D	drag coefficient
C_m	pitching moment coefficient

2. Vortex Motions

2.1. Isolated Vortex Lines

In two-dimensional flow, a "vortex" is commonly associated with a characteristic flow pattern in which the streamlines are concentric circles. There are,

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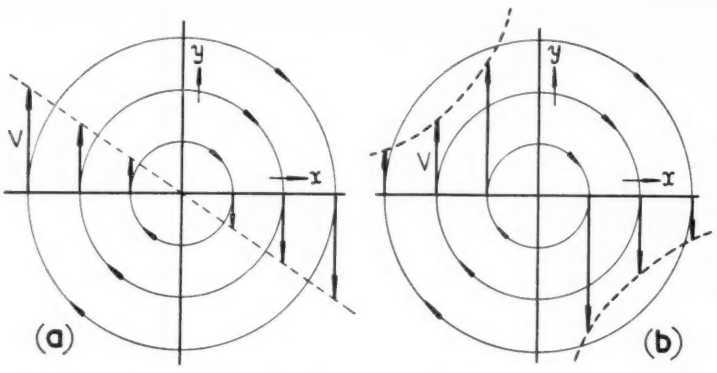

FIGURE 1 (*a* & *b*). Different types of circular flow.

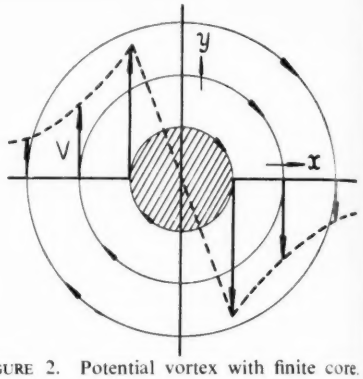

FIGURE 2. Potential vortex with finite core.

however, two entirely different types of flow which show circular streamlines (Fig. 1):—

(*a*) the rotation of the fluid as a solid body, with the velocity increasing in proportion to the radius;

(*b*) the potential vortex, with the velocity proportional to the inverse of the radius.

The flow in case (*a*) can only be caused by friction forces acting throughout the rotating mass of air; whereas in case (*b*) the decrease of pressure towards the centre precisely counterbalances the centrifugal forces acting on the fluid particles because of their circular motion. In the aerodynamics of wings, the flow is essentially inviscid except in the boundary layer and close to singular points or surfaces; hence, circular motions as in case (*a*) are unlikely to occur.

To be more precise, the two types of flow differ by the values obtained for the "vorticity" of the flow

$$2\omega(x,y)=\frac{\partial V_y}{\partial x}-\frac{\partial V_x}{\partial y}$$

and by the values obtained for the circulation

$$\Gamma=\oint(V_x\,dx-V_y\,dy)=\int\int\left(\frac{\partial V_y}{\partial x}-\frac{\partial V_x}{\partial y}\right)dxdy.$$

V_x and V_y are the components of the velocity V in the directions of two rectangular axes, x and y. In case (*a*), the vorticity is different from zero at any point in the whole flow plane; in fact, the vorticity per unit area is the same everywhere and equal to twice the angular velocity of the fluid. In case (*b*), the vorticity is zero everywhere with the exception of the core of the vortex at $x=y=0$ and thus a velocity potential exists everywhere outside the core. In case (*a*), the circulation can have any value, depending on the path of integration taken, whereas in case (*b*), for any path surrounding the core,

$$\Gamma=2\pi rV,$$

which is constant since $V=\text{constant}/r$.

The circular motion of case (*b*) is, therefore, called a "potential vortex," and it is this type of flow which demands attention. For brevity, the term "vortex" will henceforth only be applied to the core of such a flow, as is usual in fluid mechanics. In a two-dimensional flow, there is a straight "vortex line" normal to the plane of flow. These vortices are governed by the well-known theorems of von Helmholtz (1858) and Lord Kelvin (1867).

In a real fluid, even if its viscosity is small, the viscous forces prevent the occurrence of an infinite velocity at the centre of a vortex; instead, there is a gradual transition from the hyperbolic velocity distribution to a linear one inside a core of finite extent, which may be thought of as rotating like a solid body (Fig. 2). The core must not, however, be considered to be the cause of the circular motion outside it, but rather as a secondary effect of it. The existence of a finite core is, therefore, of no significance in the present context. The potential flow outside the core is produced by pressure forces, and "vortex flow" is only a name for a particular kind of potential flow. To "materialise" its core is not necessary.

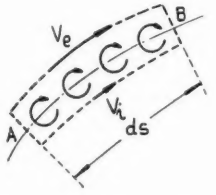

FIGURE 3. Section of a vortex sheet.

Mathematically, the vortex represents a singularity where the potential function is not regular. It is, therefore, convenient for calculation purposes to work backwards and to derive the flow field from a distribution of such singularities, or vortices of certain strengths, thus deducing the cause from the effect.

2.2. VORTEX SHEETS

In practice, vorticity rarely occurs at isolated lines but is usually distributed continuously over surfaces, as vortex sheets. In fact, isolated lines normally arise from a gradual rolling-up of vortex sheets, in contracting spirals, as shown by A. Betz[2].

Consider the boundary between two potential flows with a discontinuity of velocity there, as in Fig. 3. If AB represents a short length ds of the cross-section of the boundary, the circulation, or total vortex strength of AB, is equal to $V_e ds - V_i ds$, or $\gamma = V_e - V_i$ per unit length. Such a surface of discontinuity dividing two regimes of flow is thus equivalent to a vortex sheet on

the boundary, the strength of which is equal to the velocity difference.

We shall have to distinguish between free boundaries in a stream, which cannot sustain any normal pressure forces on them, and fixed boundaries on solid bodies, where normal pressure forces can be maintained. We shall also have to distinguish between a homogeneous flow where the total energy is the same everywhere and a non-homogeneous flow where the total head varies. Consider the incompressible flow with a boundary as in Fig. 3. The total head on either side of the boundary is

$$H_e = p_e + \tfrac{1}{2}\rho V_e^2 \quad \text{and} \quad H_i = p_i + \tfrac{1}{2}\rho V_i^2.$$

Hence, for the pressure difference,

$$\Delta p = p_i - p_e = \tfrac{1}{2}\rho (V_e^2 - V_i^2) + H_i - H_e$$
$$= \tfrac{1}{2}\rho (V_e + V_i)(V_e - V_i) + \Delta H.$$

The term $\tfrac{1}{2}(V_e + V_i) = V_b$

can be interpreted as the velocity in the boundary so that, with $\gamma = V_e - V_i$

$$\Delta p = \rho V_b \gamma + \Delta H.$$

The four cases of practical interest to be distinguished are: —

(*i*) The fixed boundary along a thin solid wall. If the total head on either side of the boundary is the same,

$$\Delta p = \rho V_b \gamma$$

and the solid wall will have to sustain this force. An example is the thin aerofoil in a free stream.

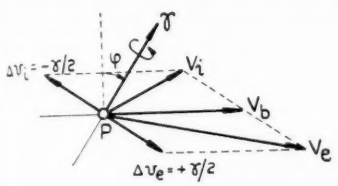

FIGURE 4. Velocity vectors on either side of a vortex sheet in the plane of the paper.

(*ii*) The fixed boundary with different total energies on either side. If the total heads on the two sides of the boundary differ by an amount not equal to $\rho V_b \gamma$, a pressure difference will exist and the solid boundary is again called upon to sustain the force. An example is the exit nozzle of a jet propulsion unit where a vortex distribution on the wall separates the two regions of different total energy and is subject to a resultant force.

(*iii*) The free boundary between regions of different total energy. If the total head on either side differs by the amount $\rho V_b \gamma$, a free boundary can be formed which takes no normal pressure force. An example is the jet of a propulsion unit with a vortex distribution on the boundary of the jet, the strength of which is related to the change of total head in the engine.

(*iv*) The free boundary between regions with the same total energy. As shown in the next paragraph, the vorticity and velocity vectors must everywhere coincide, if the total head is the same on either side of the vortex sheet. Otherwise, the sheet would deform under the pressure differences exerted by the fluid. An example is the trailing vortex sheet behind a wing of finite aspect ratio.

Case (*iv*) occurs only in three-dimensional flow. Consider then a vortex sheet in which the vorticity vector at a point P need not be at right angles to the main flow but is displaced by an angle φ, Fig. 4. In this figure, the vortex sheet is in the plane of the paper, separating a region with total head H_e and velocity V_e on top from a region with total head H_i and velocity V_i underneath. The velocity increments perpendicular to the vorticity vector are $\pm\gamma/2$ and Fig. 4 shows that

$$V_i^2 = \left(V_b - \frac{\gamma}{2}\cos\varphi\right)^2 + \left(\frac{\gamma}{2}\sin\varphi\right)^2$$

and $$V_e^2 = \left(V_b + \frac{\gamma}{2}\cos\varphi\right)^2 + \left(\frac{\gamma}{2}\sin\varphi\right)^2$$

so that $V_e^2 - V_i^2 = 2\gamma V_b \cos\varphi$.

Since the energy equations still hold,

$$\Delta p = \rho V_b \gamma \cos\varphi + \Delta H.$$

This relation may be called the Kutta-Joukowsky theorem for non-homogeneous flow. It shows that a free boundary with $\Delta p = 0$ can exist with the vorticity vector at a non-zero angle to the flow if $\Delta H = -\rho V_b \gamma \cos\varphi$. Otherwise, in the special case $\Delta H = 0$, $\varphi = \pi/2$, *i.e.* the vorticity vector lies in the same direction as the vector mean of V_e and V_i, *i.e.* V_b, which excludes the velocity increments contributed by the vortex itself.

3. Various Types of Flow past a Two-dimensional Aerofoil

3.1. INVISCID FLOW

This type of flow is well-known and various theories have been developed for calculating the flow past the aerofoil by continuing it inside the aerofoil by means of a bound vortex distribution. One might consider a distribution of vortex filaments over the whole surface of the aerofoil; or, assuming the aerofoil to be thin, the vortices may be taken as concentrated on the chord line, as is done in most theories. The shape and incidence of the aerofoil surface, which must be a streamline, determine the strength and distribution of the vortices, and from these the forces and moments acting on the aerofoil can be obtained.

3.2. VISCOUS FLOW

In the real flow past an aerofoil the vorticity is strictly different from zero everywhere. Accepting the suppositions of boundary layer theory, the vorticity may be considered to be concentrated near the surface of the aerofoil. Furthermore, if the boundary layer is thin compared with the dimensions of the aerofoil, the vorticity is concentrated into a thin vortex sheet at the surface of the aerofoil. The vorticity must be so distributed over the surface as to represent the drop from the velocity outside the boundary layer to zero at the wall and, at the same time, to make the given shape into a stream surface, *i.e.* the velocity component normal to the wall must disappear also. This is, in fact, the

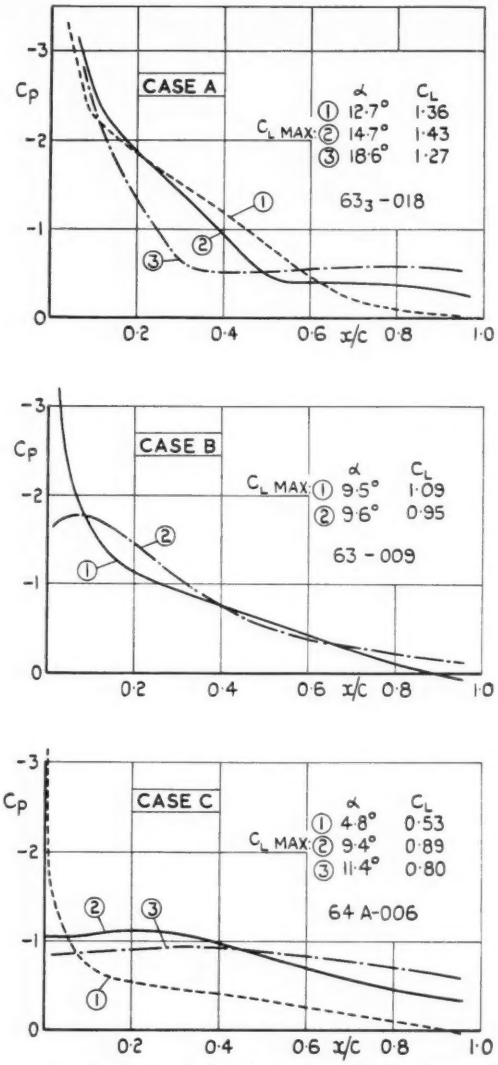

FIGURE 5. Pressure distributions on various symmetrical aerofoil sections. N.A.C.A. tests at $R=6\times10^6$, from Ref. 3.

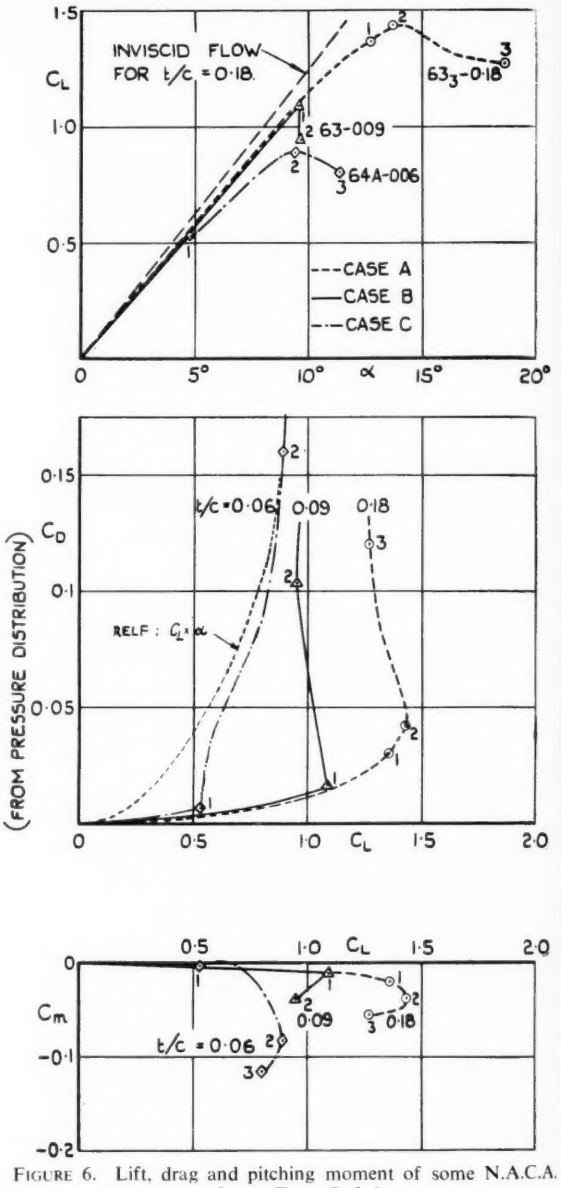

FIGURE 6. Lift, drag and pitching moment of some N.A.C.A. sections. From Ref. 3.

physical justification for using such distributions of bound vortices in the theory of aerofoils in inviscid flow.

The boundary layer on the upper surface of an aerofoil at incidence is normally thicker than that on the lower surface. However, the net vorticity in the wake behind the aerofoil must be zero, since a non-zero circulation is obtained only on a path surrounding the aerofoil. Therefore, the vorticity shed from the upper surface is equal, but of opposite sign, to that shed from the lower surface, as shown by G. I. Taylor. This implies that the dividing streamline in the wake between the air that has passed above the aerofoil and the air that has passed underneath is displaced upwards, as compared with the dividing streamline in inviscid flow which begins at the trailing edge. Hence, the overall circulation and thus the lift are reduced by the presence of the boundary layer (Preston's method). To a first approximation, this can evidently be taken account of by reducing the strength of the vortex distribution over the aerofoil and by rearranging it.

3.3. FLOW WITH SEPARATION FROM THE REAR OF THE SECTION

Whereas the trailing edge represents a stagnation point in inviscid flow, the full stagnation pressure is never reached in a real flow; instead there is a separation of the flow just upstream of the trailing edge. This may be so small, that its effect can hardly be noticed, but on relatively thick aerofoils, like that with 18 per cent. thickness-chord ratio in Figs. 5-7 (case *A*), the separation

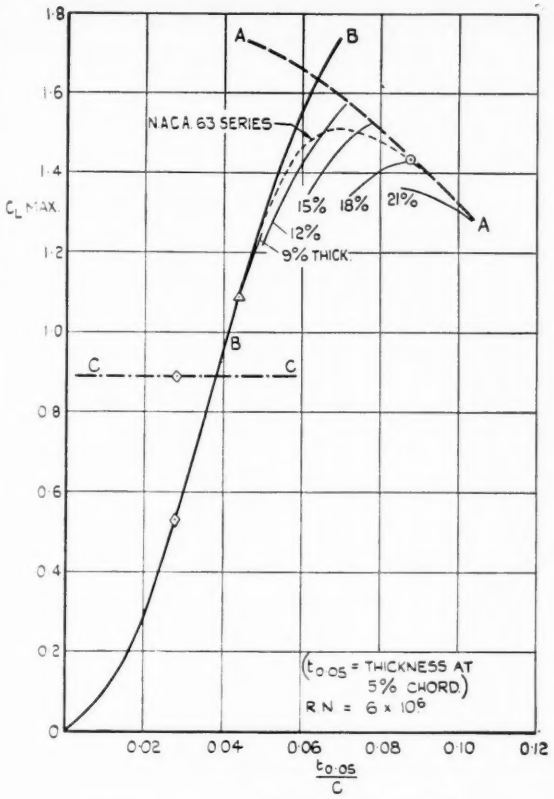

FIGURE 7. $C_{L\max}$ of some N.A.C.A. sections. From Refs. 3 and 4.

point moves forward as the lift increases—as a consequence of the increased adverse pressure gradient over the rear of the section. In the simplified picture of Fig. 8, it is imagined that a free boundary springs from the surface of the aerofoil at the point of separation. This separates a dead-air region adjacent to the aerofoil from the outer flow. The free boundary represents a vortex sheet of the third kind, the total head being different on either side of it and no normal pressure force acting on it. If the air were strictly at rest inside this "bubble," the static pressure would be constant there and hence also along the vortex sheet and along that part of the aerofoil surface which is in contact with the dead-air region. In a real flow, turbulent mixing will induce some small velocity in the bubble and may cause it to break up completely at its rear end, but the general trend is found to be well confirmed by the experimental pressure distribution in Fig. 5, case *A*.

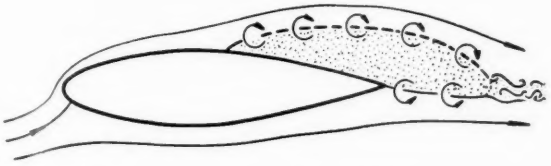

FIGURE 8. Flow with rear separation.

This type of flow requires the existence of a vortex sheet which is detached from the aerofoil surface. Its strength and position (*i.e.* the shape of the bubble) is not known and must be determined from the conditions that the boundary is a streamline and that the pressure within it is constant (*see* for example reference 5, chapter 3). This problem is, of course, considerably more complicated than that of determining the vortex strength, and hence the load, on an aerofoil in inviscid flow where at least the position of the vortex distribution is known. What can be said at once is that the curvature of the vortex sheet must increase as the pressure inside decreases. This occurs when the separation point moves forward with increasing incidence. At the same time, the more highly curved bubble will be shorter in a streamwise direction. However, all the bubbles in case *A*, as shown in Fig. 5, obviously extend into the wake behind the aerofoil.

3.4. FLOW WITH SEPARATION BUBBLES NEAR THE LEADING EDGE

The existence of a flow-separation from the rear of the section, as already described, implies that the local pressure rise which might occur near the nose of the aerofoil is not large enough to cause a separation of the boundary layer there. Aerofoil sections of moderate thickness-chord ratio like that with $t/c = 0{\cdot}09$, case *B*, in Figs. 5-7, however, possess pronounced suction peaks near the leading edge when at incidence, as illustrated in Fig. 9. This may lead to a separation of the laminar layer and, under certain conditions which have yet to be investigated, the flow re-attaches as a turbulent layer downstream. Such bubbles have already been described by Melvill Jones[1]. In some cases, a "short," or "tight," bubble is thus formed; it is of very small chordwise extent, Fig. 9, and the flow may be envisaged as sketched in Fig. 10 with a vortex sheet of the third kind, dividing a dead-air region from the outer flow. The pressure

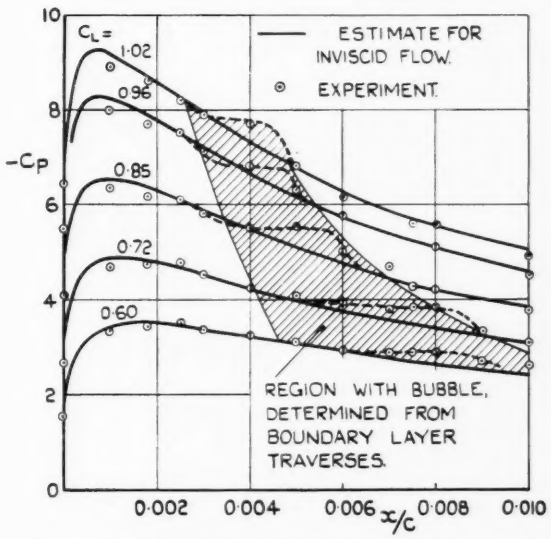

FIGURE 9. Pressure distribution near the leading edge of N.A.C.A. 63009 section. From Ref. 3.

along the bubble contour and along the part of the aerofoil surface covered by the bubble will then be nearly constant if the air inside is nearly at rest. This is illustrated by the pressure measurements in Fig. 9. Since the height of this bubble is very small, not exceeding 0·0001 times the wing chord in the example of Fig. 9 and case *B* in Figs. 5-7, the vortex distribution on the bubble surface is not very different in strength or position from the one on the aerofoil surface in the absence of the bubble, and thus the surface pressures on the aerofoil appear to differ only by a negligible amount from those calculated for non-viscous flow, Fig. 9. As a consequence, the forces and moment on the aerofoil do not betray the existence of the bubble. In case *B* of Fig. 6, a short bubble of this kind has been observed in the C_L-range between about 0·48 and 1·02.

This type of flow does *not* require a reconsideration of the chordwise distribution of vorticity, simply because the existing free boundary is insignificant in size.

On thin aerofoil sections like that with $t/c = 0{\cdot}06$, case *C*, in Figs. 5-7, a bubble is again formed as a consequence of the separation of the laminar boundary layer. But the chordwise extent of this bubble is very much greater than in the case discussed above. This is evident from the measured pressure distributions in Fig. 5 where, for instance, at $C_L = 0{\cdot}89$ the bubble extends over about 30 per cent. of the wing chord. A physical explanation of this phenomenon and a method of predicting the conditions under which a short or a long bubble will occur are required. Since the pressure inside the bubble is again very nearly constant, these "long" bubbles can be visualised as dead-air regions bounded by the wing surface and by a vortex sheet separating it from the main potential flow. This vortex sheet is much less curved than those of the short bubbles and the pressure inside it is more nearly equal to the free stream pressure. Since the bubble is comparable in size to the aerofoil, the pressure field of the flow about the aerofoil with bubble is appreciably different from that of the flow past the aerofoil without bubble. It is also significant that the occurrance of such long bubbles is accompanied by an appreciable drag force.

On the aerofoil *C* in Figs. 5 and 6, a short bubble of the first type has been observed at low C_L-values between about 0·35 and 0·53. In the C_L-range above that the bubble is of the long type. The length of the latter increases as the incidence increases, whereas the short bubble type decreases in length with increasing incidence as can be seen from Fig. 9. The long bubbles are, therefore, sometimes called "expanding" bubbles. Fig. 6 shows that the drag of an aerofoil with a long bubble follows closely the relation $C_D = C_L \times \alpha$, which implies that the suction force at the leading edge has been lost (*see* E. F. Relf[6]).

It is essential with this type of long bubble to take account of the change which it causes to the strength and distribution of the bound vorticity. The problem is somewhate similar to that produced by a rear separation as described in Section 3.3, the essential difference being that the long bubble described here begins and seems to end within the wing chord as a consequence

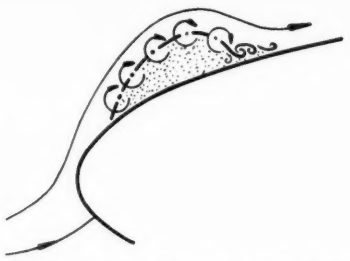

FIGURE 10. Flow with short bubble near leading edge.

of a turbulent mixing process which leads to what may be called a reattachment of the flow. It appears that this problem has not yet been treated at all.

3.5. WHOLLY SEPARATED FLOW

It is difficult to state precisely when an aerofoil is considered to be stalled. It may be defined here as a flow where the stagnation streamline follows the upper side of the aerofoil contour for a very short length only and then separates from the surface so that practically none of the upper surface of the aerofoil is "wetted" by the main air stream. One such flow pattern would incorporate a big dead-air region bounded by the upper surface, a free boundary springing from the point where the stagnation streamline separates from the surface, and another free boundary originating from the trailing edge, as indicated in Fig. 11.

The static pressure inside the separated region and hence along the upper surface of the aerofoil is then constant. If it is equal to the free-stream pressure, the dead-air region extends to infinity downstream and flow is related to the well-known Kirchhoff flow past a plate normal to the stream. It appears, however, that the pressure inside is normally slightly less than that of the free stream, the stagnation streamline following the contour into a region of reduced pressure before it separates. The dead-air region is then of finite length. Under the effect of large-scale turbulent mixing such large dead-air regions may be unstable, in which case the free boundary breaks down, resulting in an irregular flow and a varying pressure along the aerofoil surface. Case *B* at $C_L = 0{\cdot}95$ in Fig. 5 provides a typical example of this type of flow. The inherently disorderly character of such a flow defies any ready analysis. What appears to be a general characteristic of this type of flow is some measure of reversed flow near the surface, resulting in a slight pressure rise from nose to trailing edge.

3.6. MAXIMUM LIFT

Corresponding to the various types of flow, the maximum values of the lift obtainable varies considerably. To illustrate this, the pressure distribution along the upper surface at which maximum lift has been found is drawn as a full line in Fig. 5, and Fig. 6 gives the complete lift, drag and pitching moment curves for three typical aerofoils representing the cases *A*, *B* and *C* as discussed already. Fig. 7 shows a summary of measured $C_{L\,max}$ values for a series of symmetrical N.A.C.A. sections plotted against a parameter which measures

the nose thickness of the aerofoil as proposed by Multhopp[4]. The use of such a parameter is, of course, reasonable only if the aerofoils are geometrically related to each other. Each measured aerofoil section provides a point in this diagram, and envelopes have been drawn marking the highest $C_{L\,max}$ values obtained. These limiting lines for $C_{L\,max}$ are significantly different for cases *A*, *B* and *C*.

Only in case *B* does an abrupt change in flow pattern coincide with maximum lift. Although a short bubble near the leading edge does not normally produce any significant change in the general type of flow, its curvature must increase as the pressure inside decreases with increasing incidence, and there seems to be a stage where the bubble becomes too curved for the flow to follow it and then re-attach to the surface. As a consequence, the bubble may "burst" and the flow become wholly separated and irregular, resulting in a *sudden* drop in lift, an increase in drag, and a nose-down change in pitching moment.

In case *A*, $C_{L\,max}$ is reached when the rear separation bubble is well developed, and *no change* in flow pattern occurs at $C_{L\,max}$. In the example given, the bubble starts at about midchord at $C_{L\,max}$. The reason for the subsequent drop in lift is only indirectly caused by the separation bubble. This itself gives more suction but the suction upstream of the bubble is reduced as a consequence of the overall flow pattern (Fig. 8).

In case *C*, $C_{L\,max}$ is again not related to the existence of either the short or the long bubble near the leading edge. $C_{L\,max}$ occurs in the middle of the long bubble regime. That a maximum value of the lift is reached is related to the fact that, although the length of the bubble—and hence the region of comparatively high suction—increases with incidence, the value of the suction inside necessarily decreases (*see* Fig. 5). It appears that, below a certain value, the nose thickness affects the creation of the initial short bubble and the subsequent changeover to a long bubble type of flow but not the actual value of $C_{L\,max}$, to a first approximation. This is another indication that $C_{L\,max}$ normally occurs well *within* a regime of non-homogeneous flow past an aerofoil with a free boundary attached to it.

It may be concluded that $C_{L\,max}$, even on two-dimensional aerofoils, cannot be predicted unless more is known about such mixed flows with fixed and free boundaries. In particular, the relation between the boundary layer and the formation of such bubbles and their stability must be studied, as well as the flow past aerofoils with full established dead-air regions.

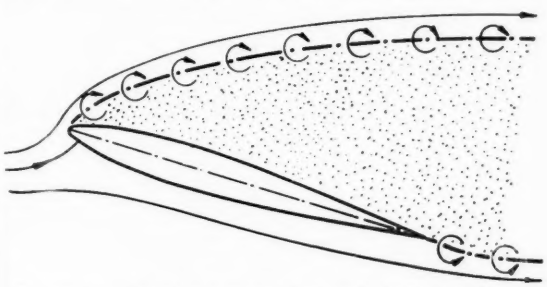

FIGURE 11. Completely stalled flow with dead-air region.

4. Three-dimensional Flow past Swept Wings

4.1. INVISCID FLOW

The various types of flow and the conditions for their existence are presumably determined by the pressure distribution along the surface of the aerofoil in inviscid flow. There is no fundamental difference between the type of pressure distribution along a two-dimensional aerofoil and that along a section of an unswept wing of finite aspect ratio, except near the tips and unless the aspect ratio becomes very small. The main differences between the two- and the three-dimensional flow consists in the well-known fact that the downwash produced by the sheet of trailing vortices alters the effective incidence of a section on a three-dimensional wing. Thus different lift coefficients may be produced at different stations along the span, and, as a consequence, the changeover from one type of flow to another may not occur simultaneously at all spanwise stations. The untwisted unswept wing of elliptic planform is distinguished from wings with other planforms in that the lift coefficient, and thus the pressure distribution, are the same at all spanwise stations, so that the two-dimensional results can be directly applied.

The characteristic effect of sweep in the planform is that, in addition to a change of incidence by the trailing vortices, the bound vortices on the wing change their geometric configuration and the whole flow field near the centre and near the wing tips differs appreciably from that around a two-dimensional section of the same shape. We may speak of an "induced camber" from the bound vortices to signify the change in chordwise loading and surface pressure distribution in the centre and tip regions. If, for instance, the planform is such as to give a constant downwash along the span from the trailing vortices, the pressure distributions at various spanwise stations are still not the same (*see* Ref. 11). For swept-back wings, the suction forces on the surface are shifted backwards in the centre region and forward in the tip regions. This makes the direct application of the results from two-dimensional aerofoils impossible. Thus, a section which normally (*i.e.* under two-dimensional flow conditions) has a rear bubble of type *A* above, might change over to a short front bubble of type *B* if near the tip of a swept-back wing, because of the increased suction peaks near the nose. Again, an aerofoil section with normal stall of type *B* might stall at a lower C_L than expected from Fig. 7, due to the tip effect; it might even change over to a flow of type *C* with a long bubble near the leading edge.

These changes in the pressure distribution due to sweep can be eliminated by the application of camber and twist. This problem is simplified considerably by the fact that, to a first approximation, only one special camber line is required to do this. It is only the amount of camber that is dependent on the angle of sweep, the spanwise position of the section, and on the value of local C_L; the shape of the camber line is always the same. This has been shown in an unpublished paper by G. G. Brebner.

In general, it may be said that, for swept wings which are not cambered and twisted in such a way as to

offset these effects, there are (in inviscid flow) chordwise pressure distributions which vary widely along the span. The same type of flow cannot be expected to be found, therefore, all along the span under all conditions; at high incidence, local dead-air regions with free boundaries must be expected to occur.

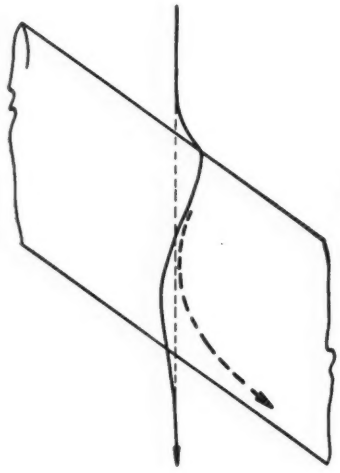

FIGURE 12. Path of a particle outside the boundary layer (full line) and inside (broken line) on a sheared wing.

4.2. VISCOUS FLOW

The development of the boundary layer along the surface of a swept wing also shows some characteristic deviations from that on the two-dimensional unswept aerofoil. The question of the transition from laminar to turbulent flow is not discussed here but the wing at lift, with the transition position known, is considered. A particle in the boundary layer is in a state of equilibrium between the inertia, pressure and viscous forces. On a sheared wing, as in Fig. 12, a particle outside the boundary layer will follow a curved path in accordance with its inertia and pressure forces. A particle inside the boundary layer, having lost part of its momentum through viscosity, may not follow a path parallel to that of a particle outside the boundary layer but one which offers a less steep adverse pressure gradient, *i.e.* the particle branches off sideways, in contrast to the flow past an unswept aerofoil of infinite span where, by definition, all particles remain in planes perpendicular to the wing.

The type of boundary layer profile obtained on a swept-back wing is illustrated in Fig. 13. This shows the velocity in the boundary layer resolved into components perpendicular (V_ξ) and parallel (V_η) to the leading edge, measured immediately behind the trailing edge. These velocity profiles are compared with an experimental boundary layer profile at the same C_L and Reynolds number on a two-dimensional unswept aerofoil having a similar chordwise pressure distribution. Transition took place very close to the leading edge in both cases. $V_{\xi\delta}$, $V_{\eta\delta}$, and V_δ are the respective values at the edge of the boundary layer. It is found that the profile of $V_\xi/V_{\xi\delta}$ is quite similar to that for a two-dimensional flow but is displaced upwards by a layer of air in which there is almost no velocity component normal to the leading edge. Only within this *sublayer* does $V_\eta/V_{\eta\delta}$ differ appreciably from the two-dimensional profile.

In the present context, this nearly uniform sublayer should not be mistaken for a sign of flow separation; this type of flow is different from that with a separation from the rear of the section where a free boundary is formed, as discussed in Section 3.3. The overall thickening of the boundary layer which accompanies the existence of the sublayer with its continuous accumulation of material as it flows outwards, however, reduces the circulation around the aerofoil section as compared with that on an unswept aerofoil at the same effective incidence.

4.3. THE TIP VORTEX SHEET

It has been seen in Section 3.5 that in the state of completely stalled flow past a two-dimensional aerofoil a vortex sheet springs from the leading edge and another from the trailing edge. The vorticity vector on these sheets is directed along the span of the wing. If the

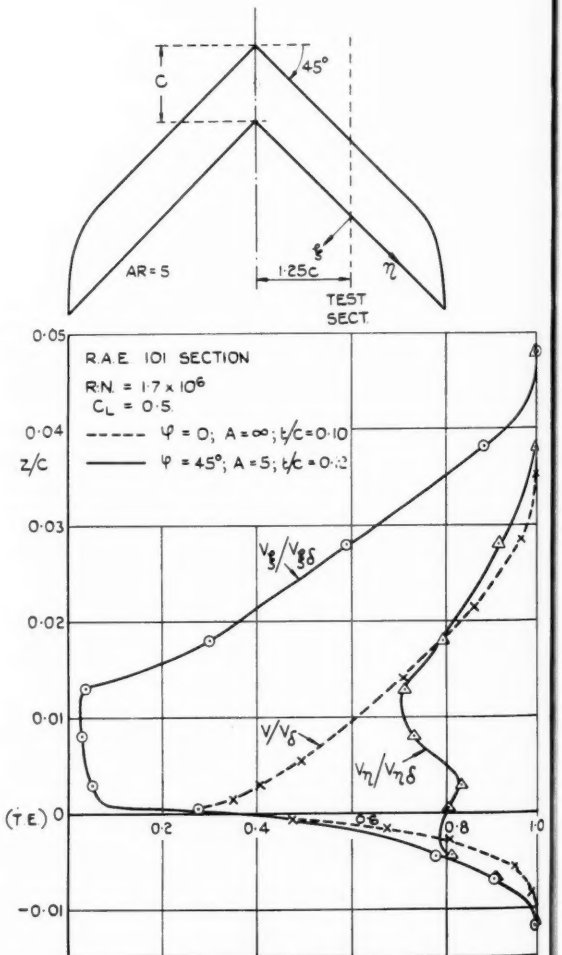

FIGURE 13. Comparison between boundary layer profiles on a straight and on a swept wing, at trailing edge.

wing is of finite span, these two vortex sheets must join up near the ends of the wing because they cannot end in the free stream (Fig. 14). The vertical part of the vortex sheet may be called the "tip vortex sheet," or briefly "tip vortex" if it is borne in mind that it is not a discrete vortex filament but a sheet (which may, however, roll up into a single vortex farther downstream). Its existence signifies that the flow not only separates from the leading edge but also along the tip chord.

If, at a smaller incidence, the flow is attached to the leading edge all along the upper surface, it does not necessarily follow that the flow should also be attached at the tip. The vortices on the upper free boundary having collapsed into the wing surface, some of the vortex filaments may still leave the wing at the tip chord and continue downstream as a sheet of trailing vortices, Fig. 15. This tip vortex sheet will persist above a certain incidence at which the flow is attached along the tip chord. The value of this incidence will depend on the load distribution on the wing and also on the tip shape, as well as on the Reynolds number.

The effect of the tip vortex has been calculated by using the fact that the whole sheet of trailing vortices in the Trefftz plane behind the wing is to a first approximation, the same as if it had been produced by solid endplates at the wing tips (*see* reference 7). This procedure is justified and suitable for calculating the downwash in the Trefftz plane. If this downwash distribution is carried forward to the wing in the usual manner, a load distribution on the wing is obtained, which does not fall to zero at the wing tip, *i.e.* there is a sideload on the "endplate" which, in fact, the tip vortex is unable to sustain. However, experience has shown that a good estimate of the load and of the pitching moment can be obtained by this "endplate method," which must imply that the real shape of the tip vortex sheet is not very different from that of a solid vertical endplate at the tip together with its wake. The method in this form is, however, restricted to obtaining the overall lift over the chord; the chordwise distribution of the lift cannot be predicted. To do this, the downwash on the wing must be worked out fully, taking into account that the trailing vortex elements on the tip vortex sheet start at different streamwise positions, as indicated in Fig. 15.

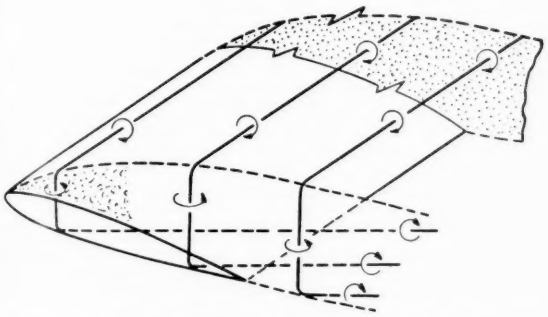

FIGURE 14. Completely stalled flow with dead-air region near the tip of a wing.

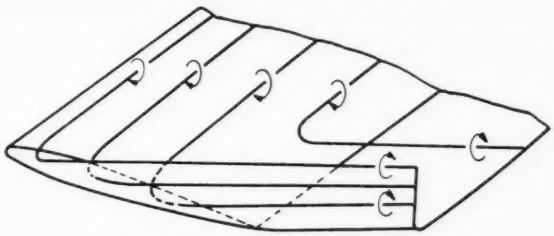

FIGURE 15. Flow with tip vortex sheet.

The main effects of the tip vortex sheet are:—

(*a*) An increase in lift near the tip which makes the lift curve non-linear (in the range from (1) to (2) on the untapered wing in Fig. 16).

(*b*) A nose-down pitching moment which produces the first characteristic kink in the pitching moment curve (at point (1)).

Since the height of the tip vortex sheet and thus the effect of this vortex must depend on the tip chord, the effect will be less noticeable on tapered wings; it disappears altogether on wings with pointed tips.

4.4. DIFFERENT TYPES OF FLOW ON THE SAME WING

Among the many possible combinations of the various types of flow described in Section 3 at different regions along the span, consider a simple one where the flow is broken down over the outer part of the wing whereas there is normal, attached, flow about the inner part of the wing. Such a flow may occur when a short bubble is first formed near the leading edge along part of the span, which subsequently breaks down near the point where the highest and farthest forward suction peaks occur, as illustrated in the example in Fig. 17. The flow may either break down completely, or a long bubble near the leading edge may be formed. In the latter case, the pressure distribution may develop as illustrated by Fig. 18 in which there is a resemblance to the flow in case *C* as discussed in Section 3.4, Fig. 5.

From experiment it is found that such a flow with two different regimes on the wing is quite stable and that the changeover appears to be almost discontinuous. This means that the pressure field is such as to produce different velocities side by side at the boundary between the two flow regimes, which is equivalent to saying that there is a vortex sheet in the sense as defined in Section 2.2, acting as a free boundary. This may be called a "part-span vortex sheet" because it originates from some point somewhere along the span, and not from the tip. Since the total head is the same on either side of the vortex sheet, the vorticity vector must lie along the mean flow direction (except in the boundary layer where it originates) *see* Fig. 19. A sheet of trailing vortices is thus formed which may stand up from the wing surface. It originates from the wing surface mainly in the front portion of the section where the pressure changes rapidly, Fig. 17, and it may be thought of as a continuation of those bound vortices (on the wing, or on the surface of the short bubble) which do not continue outboard, Fig. 19.

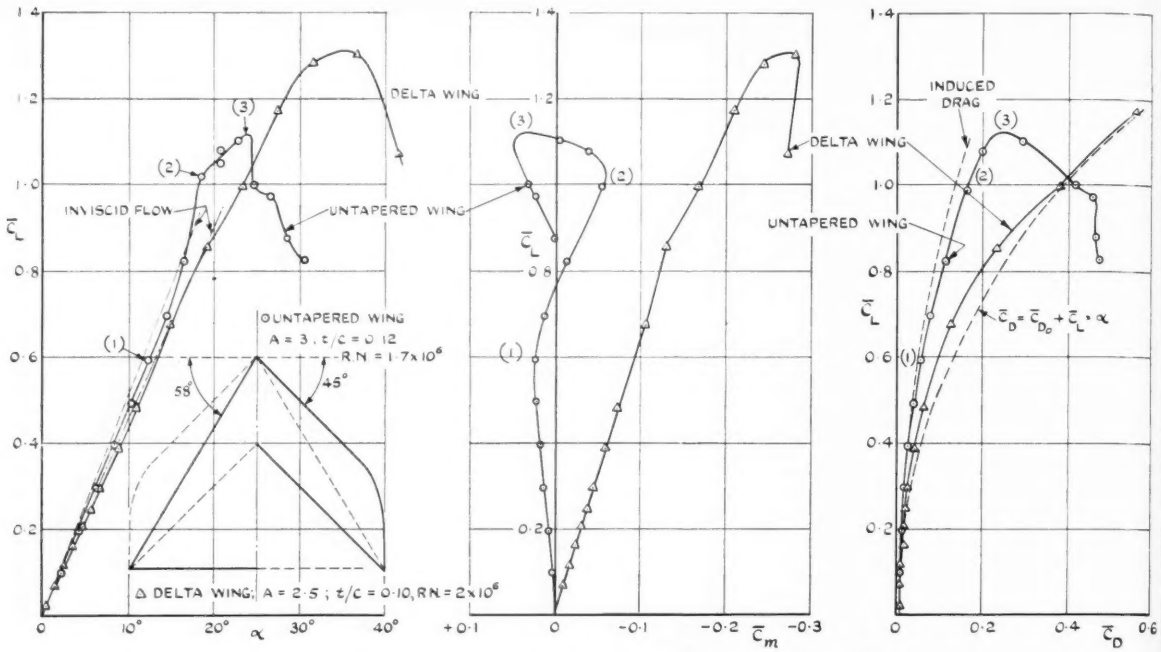

FIGURE 16. Overall forces and moments on a delta wing (Ref. 10) and on an untapered wing (Ref. 8).

Behind the wing, the part-span vortex sheets may join the sheet of ordinary trailing vortices; in other cases, the part span vortex sheets may be detached from, and above, the sheet of trailing vortices. Both of them may eventually roll up forming two pairs of discrete vortex filaments along the free stream in the wake far behind the wing.

Little is known as yet about the conditions which lead to the formation of these part-span vortex sheets. A study of the flow in the junction of two vortex sheets is needed to find out more about the details of how the part-span vortex sheet originates and how it is shaped.

The part-span vortex sheet is of interest mainly in three respects:—

(*i*) It is necessarily associated with a change in the downwash field over the wing and thus with a modification to the load distribution.

(*ii*) It assumes a characteristic shape in order to fulfil the condition that it cannot sustain any force.

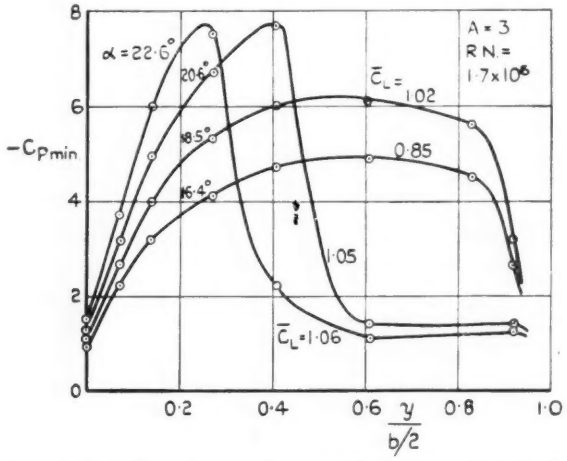

FIGURE 17. Minimum pressure near L.E. on an untapered 45° swept-back wing. From Ref. 8.

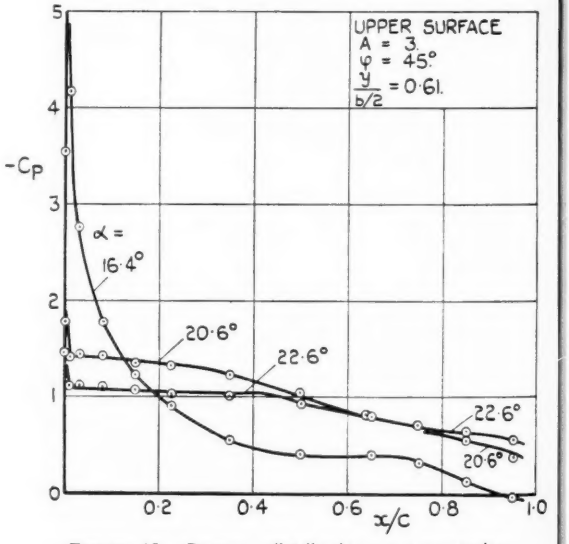

FIGURE 18. Pressure distribution at a spanwise station of the wing of Fig. 17.

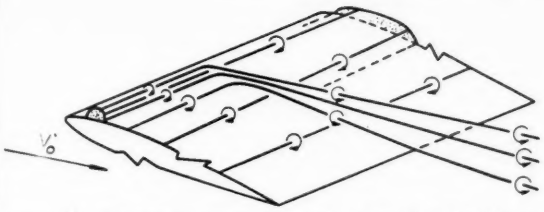

FIGURE 19. Part-span vortex sheet with short L.E. bubble on one side and a long L.E. bubble on the other.

(*iii*) It changes the downwash over the tail surface of the aircraft.

We are also interested in its development and movement with changing incidence and how far it stabilises the mixed flow.

To estimate the change in the downwash, the shape of the vortex sheet must be known. If its trace in the Trefftz plane consists of two vertical lines above the horizontal line of the ordinary trailing vortices, Fig. 20, similar to those produced by two vertical solid part-span plates on the wing, and if these traces were straight behind the points at which the vortex sheets originate on the wing (*i.e.* if the vortex sheets were plane surfaces), an increase of the load on the wing between them and a reduction of the load outside them would follow from the modified downwash field, in the same manner as such a change of loading occurs with solid plates. This is not possible because the vortex sheets would then have to sustain side forces. The vortex sheets must, therefore, assume the shape of curved surfaces, as sketched in Fig. 20. The curvature will be such as to give no side force. Part-span vortex sheets thus behave differently from the ordinary system of bound and trailing vortices. The latter follow the wing surface, because this can sustain a force; but behind the wing, the existence of a non-zero downwash means that the sheet of trailing vortices moves downwards. Vertical part-span vortex sheets, on the other hand, deform on the wing, where they are created, but they will keep their spanwise position behind the wing because, by then, the sidewash which they themselves induce is compensated by the sidewash induced by the horizontal sheet of trailing vortices. These conditions require further investigation.

Since the vortex sheets are also stream surfaces, curved sheets imply curved flow near them, and it is along these curved streamlines that the downwash distribution in the Trefftz plane could be carried forward to the wing. Also, in analysing measured pressure distributions on the wing, only the circulation obtained by integrating the velocity along such curved streamlines can be expected to give values which can be interpreted in the usual terms. This implies that the "streamwise" rather and the "chordwise," load distribution should be evaluated, especially in cases where a section y = constant cuts across the part-span vortex sheet. This is illustrated by the examples in Fig. 21 where the spanwise lift distributions as obtained from an integration of

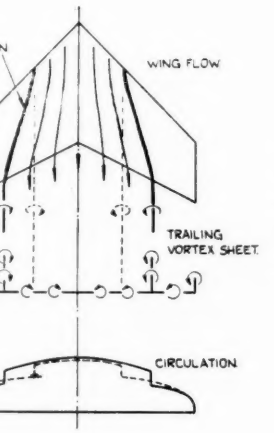

FIGURE 20. Part-span vortex sheets and span loading curves (circulation) for part-span plates (broken line) and vortex sheets (full line).

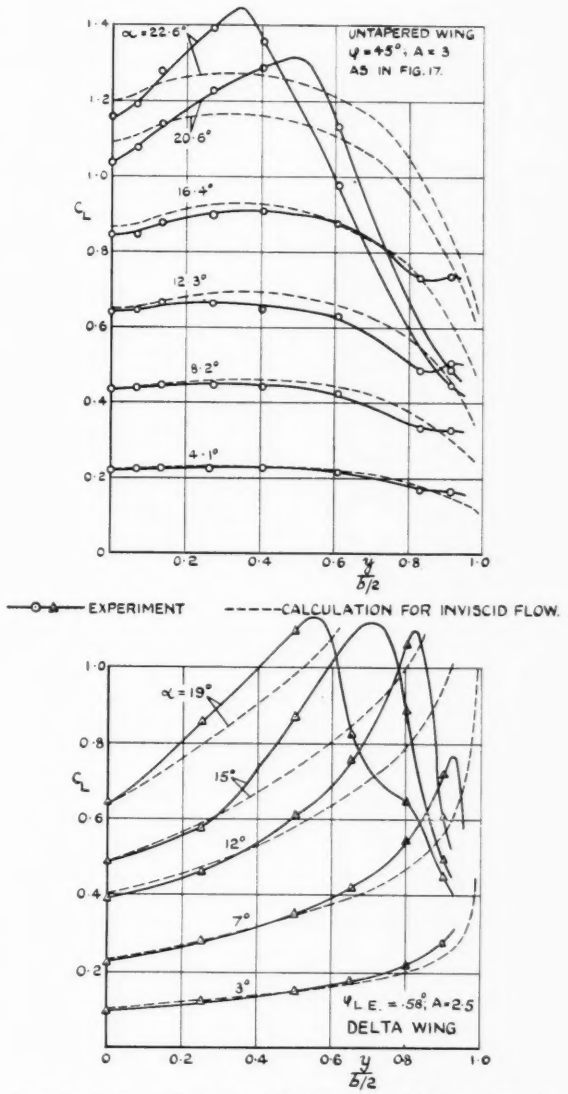

FIGURE 21. Spanwise lift distributions on a delta wing (Ref. 9) and on an untapered wing (Ref. 8).

the pressures along the chord, are shown to be very much smoother than might be expected from the almost discontinuous changes of the leading edge suction peaks shown in Fig. 17.

As the incidence increases, the height of the part-span vortex sheets and hence the lift on the portion of wing between the vortex sheets increases too, the latter possibly more than linearly, and the conditions which led to the initial breakdown of the short leading edge bubble at some point will now occur farther inboard. This means that the two vortex sheets on each half wing approach each other with increasing incidence. This inward movement of the part-span vortex sheets can be seen from the example in Fig. 17. On swept-back wings, the suction peaks in the centre region are very much less and occur farther back along the chord than at spanwise stations farther out on the wing. It is not likely that there will be a short bubble from laminar separation in the centre region when there is one farther outboard, and thus there is little chance that the part-span vortex sheets will move right into the centre (or into the wing-body junction where similar conditions prevail). This implies that the inward movement of the vortex sheets is likely to stop at a short distance from the centre of the wing. The final breakdown will probably be caused by a flow separation from the rear of the centre section.

The presence of the part-span vortex sheets must affect the downwash at the tail. To put it crudely, some of the bound vortices, which, in the absence of this sheet, would run along the wing nearly up to the wing tips, now leave the wing by way of these vortex sheets much farther inboard and thus pass much closer to the tailplane. This implies an increase of the downwash at the tail, which coincides with the nose-up tendency of the pitching moment on the wing. Bearing in mind that swept-back wings normally have a span-loading distribution which leads to a comparatively small downwash at the tail, or even to an upwash if the aspect ratio if the wing is large, the presence of the part-span vortex sheets may greatly affect, or even reverse the sign of, the vertical velocity component at the tailplane. It must be beneficial in this respect to have the tailplane in a low position so that it moves away from the part-span vortex sheets as the incidence increases.

Above all, the presence of the part-span vortex sheets affects the pitching moment of the wing. If the loss of lift occurs on the outer parts of the wing surface, an unstable, nose-up, change of pitching moment must result. This may be very abrupt. Considering the change of flow pattern as the incidence increases, it can be imagined that the breakdown of the flow first occurs near the spanwise station at which the suction peaks are highest and farthest forward. In the initial stages, the existence may be imagined of a narrow strip of broken-down flow which is bounded on either side by part-span vortex sheets of opposite circulation, the flow in the immediate neighbourhood of the wing tips still being attached to the wing surface everywhere. This appears to be an unstable state and the two outer vortex sheets will move rapidly towards the tips as the incidence increases, thereby reducing the strength of the tip vortex sheets, whereas the two inner vortex sheets will move inwards but much more slowly. In this way, the lift near the wing tips and hence the pitching moment undergo rapid changes in the initial stages. In the further development of the part-span vortex sheets, their inward movement, steadily enlarging the regions with broken-down flow, must continue to produce unstable pitching moments. This will stop only if the inward movement of the vortex sheets can be arrested.

The lift and pitching moment changes due to the tip and part-span vortex sheets on an untapered 45° swept-back wing are illustrated in Fig. 16. The spanwise lift distribution is shown in Fig. 21. The lift increase accompanied by a stable change of pitching moment at

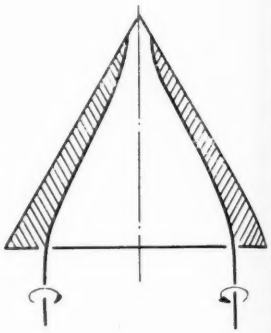

FIGURE 22. Possible shape of part-span vortices on a delta wing.

point (1) caused by the tip vortex sheets can be seen clearly, as well as the decrease in lift slope accompanied by an unstable change of pitching moment at point (2), caused by the part-span vortex sheets. Maximum lift is reached at point (3) somewhere in the middle of the "part-span vortex regime" when the main load-carrying part of the wing between the part-span vortex sheets has been so curtailed that it cannot counterbalance the loss of lift outboard. $C_{L\,max}$ does not necessarily mean that the part-span vortex sheets have reached the centre of the wing, or that the flow has completely broken down. Any change in the Reynolds number of the flow can be expected to affect the points at which transitions from one type of flow to another occur; it is less likely to affect the overall picture of the flow.

The thickness-chord ratio of the wing section affects mainly the values of the lift at which the various changes occur, these being lower the thinner the wing. In the first place, the tip vortex will occur at a lower $\bar{C}_L$; secondly, the onset of the part-span vortices will occur earlier. Thus both points (1) and (2) will be lower, and it can be expected that the tip-vortex regime from (1) to (2) will cover a smaller $\bar{C}_L$-range, whereas the part-span vortex regime from (2) to $C_{L\,max}$, point (3), and beyond will cover a larger $\bar{C}_L$-range on a thin wing than on a thick wing. Once the outer wing has stalled, the flow there will be little affected by the thickness-chord ratio of the aerofoil. Since, for wings of moderate and small thickness-chord ratio, $C_{L\,max}$ occurs in the middle of the part-span vortex regime, the actual value of $C_{L\,max}$ need not be greatly affected by the wing thickness.

The wing plan form affects in the first place the loading and pressure distribution of the wing in normal flow and thus the overall lift value and the spanwise position of the initial part-span vortex sheets. Wings with pointed tips, for example, cannot have a tip vortex, but a part-span vortex sheet, albeit very small, will exist as soon as the incidence is different from zero. Because of the particular shape of the part-span vortex sheets, Fig. 22, delta wings with highly swept-back leading edges need not lose much lift even if the vortex sheets have moved almost into the centre of the wing and, as a consequence, the change in pitching moment will be less violent than on a less tapered wing. This is illustrated by the comparisons given in Figs. 16 and 21 between the results for an untapered 45° swept-back wing and those for a delta wing with 58° leading-edge sweep, from tests by Orlik-Rückemann[9] and Berndt[10]. The spanwise lift distributions in Fig. 21 follow closely the curves calculated by a method similar to that in Ref. 11, at low C_L-values. On the untapered wing, the lift reduction due to boundary layer thickening and the effect of the tip vortex sheets are evident. On the delta wing, the theoretically infinitely high C_L-values at the tips are, of course, never reached in practice, and the effects of part-span vortex sheets can clearly be seen at all C_L-values; the increase of lift inboard of the vortex sheet is quite spectacular, and it will be seen that the vortex sheet evidently moves inward as the incidence is increased. The same effect of part-span vortex sheets is observed on the untapered wing, with similar deviations from the curves calculated for inviscid flow in the absence of such vortex sheets; but the part-span vortex sheets set in at much higher incidences than for the delta wings*. In view of this, it is remarkable how little the lift and pitching moment characteristics of the delta wing seem to be affected by the ever-present part-span vortex sheets; whereas the untapered wing reacts violently to the onset of both the tip vortex and part-span vortex sheets. This must be related to the geometrical differences between the two wings. The delta wing, however, loses most of the leading-edge suction force and its drag approaches Relf's curve, although this wing was not particularly thin ($t/c=0{\cdot}10$). At high incidences, the part-span vortex sheets on a delta wing can be expected to be very stable and flight conditions may possibly be encountered, which may be as undesirable as the flat spin was on conventional aircraft. Moreover, most of the lift is then concentrated on a narrow area between the vortex sheets, with a comparatively large portion of it at the rear, so that any disturbance which leads to a rear separation might possibly cause a violent change in longitudinal stability.

*In this context the difficulty encountered when comparing measured and calculated spanwise lift distributions may be noted. Very often we find the too readily drawn conclusion that some particular theory has been confirmed by experiment when, in reality, any calculation based on wholly attached inviscid flow should *not* agree with experiment because of the presence of the effects of viscosity and of tip and part-span vortex sheets.

4.5. OTHER POSSIBLE FLOW COMBINATIONS

There is a multitude of possible ways in which the various types of flow discussed in Section 3 may be combined on a wing of finite span and little is known as yet about which combinations of practical interest occur most frequently and what determines their stability.

For relatively thin aerofoil sections, the co-existence of regimes with short and long dead-air bubbles may be expected. The changes in the spanwise load distribution are not necessarily as drastic in such a case as in the case assumed in the previous section. In fact, the sectional lift slope at a spanwise station with a long leading edge bubble may be very nearly the same as that of a section with a short leading edge bubble, *see* Fig. 6. There may thus be cases where no trailing vortices of appreciable strength leave the wing from the boundary between the two regimes. There will be little change in downwash at the tail in such a case. However, there will be streamwise vortices on, and possibly above, the wing surface at the boundary because the chordwise loadings are different in the two regimes. Thus a vortex sheet may be visualised which originates near the leading edge, only to disappear again somewhere downstream within the wing chord. In other words, bound vortices near the leading edge on the inner part of the wing will turn round into the stream direction at the boundary between the two regimes, and subsequently turn back again to the original direction and continue as bound vortices farther back on the outer part of the wing. It then depends on the change in downwash produced by these streamwise vortices whether the spanwise loading will be affected or not. The change in chordwise loading will cause a change in pitching moment but this will be much less violent than in the case discussed in the preceding Section. It appears that really big changes in pitching moment will occur only when the sectional lift slope on the outer parts of the wing drops appreciably, for example when the flow there breaks down completely, or when the downwash from the streamwise and trailing vortices changes considerably. This may happen with long bubbles at a later stage, and it is quite possible that there are three different regimes along the span:—Normal flow, or flow with a short bubble, in the inner portion; flow with a long bubble in the regions adjacent to this, and a completely stalled flow in the tip regions; the latter two regions being separated from one another by another pair of part-span vortex sheets in much the same manner as described in Section 4.4. Such a case may happen with model tests at very low Reynolds numbers, below 10^6, say.

Another possible flow combination of practical interest consists of a normal unstalled flow inboard with regions of rear separation outboard near the wing tips. If the chordwise pressure distributions in adjoining sections in the two regimes are like those in Fig. 5, very little shedding of vortices into the wake may occur, bound vortices from the front part of the unstalled region continuing mainly on the free boundary of the rear separation bubble.

A similar flow, with the order of the regimes reversed, may occur if there is a rear separation in the junction between a wing and a fuselage, spreading out some way along the span and followed farther out by normal unstalled flow. In that case, vortices on the free boundary of the separation bubble may run first upstream and then curve round and continue as bound vortices along the span in the unstalled regime, Fig. 23. However, the overall lift along sections in the region with separation may be less than that farther out, in which case some trailing vorticity would be left behind. The direction of this vorticity as a vector is then opposite to that of the ordinary trailing vortices. This is important because such vortices produce an upwash at the position of the tailplane.

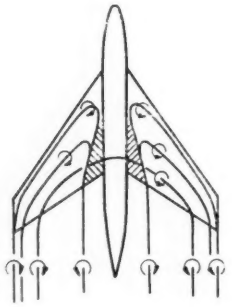

FIGURE 23. Vortex system with rear separation in the wing-body junction.

In discussing the various flow combinations the assumption has always been made that there is an abrupt change from one flow regime to the other somewhere along the span. Although almost discontinuous changes have been observed in experiments, such changes could be gradual.

4.6 EFFECT OF FENCES

There are various means by which one could try to influence and to control the development of these vortex systems. The wing could be designed by means of the appropriate plan form, twist, camber, and section shape, so as to prescribe the position where the first breakdown occurs and also, to a certain extent, the manner in which it takes place. Alternatively, one may try to *fix the location* of the vortex sheets by some auxiliary device such as fences, or leading-edge fillets or nose droop in the wing root. The particular case of fences may serve as an illustration.

Historically, fences were thought of as a means of obstructing the outward drift of boundary layer material. This may still be the main effect of fences over the rear part of a section on swept-back wings of large aspect ratio, and their application may be beneficial if this boundary layer drift would otherwise lead to a rear separation of the flow. There are, however, other possible effects of a fence.

Consider the effect of a large plate across a swept-back wing in a streamwise direction. This acts as a reflection plate to the bound vortices on either side and it can sustain a side force. On the inboard side, the flow conditions resemble those at the centre of a swept-forward wing, and on the outboard side, they are like those at the centre of a swept-back wing. On the inboard side, the suction peaks are increased and occur farther forward than on the wing without plate, and on the outboard side, they are smaller and occur farther back, as illustrated by the isobar pattern in Fig. 24. Firstly, this affects the chordwise loading and hence the sectional lift slope, which is higher on the inboard side and lower on the outboard side than on the undisturbed wing. Secondly, as a result of the different chordwise loadings on either side, some of the bound vortices from the inboard side will run up the plate (which will thus carry a side force) and subsequently leave the plate as a vertical sheet of trailing vortices. The wake in the Trefftz plane thus has a very similar shape to that of a wing with a part-span vortex sheet, Fig. 20. This affects the downwash from the trailing vortices and thus the loading on the wing, and also the downwash at the tailplane.

Fences as applied in practice are not so large as to produce a full reflection of the bound vortices, and the change in downwash from the trailing vortices need not be considerable. Nevertheless, the tendency must be to increase the lift inboard and to decrease it outboard of the fence. This means in the first place that a fence can, for a given overall lift, shift the position where the highest local C_L occurs to a position farther inboard and, because the span loading curve may become more uniform, the peak C_L-value for a given overall $\bar{C}_L$ will be reduced as compared with that on the wing without fences. For wings with moderate sweepback and taper, the application of several small fences in the appropriate spanwise positions may be sufficient to convert the span-loading into one where the highest C_L occurs near the middle of the wing, and so avoid tip stalling altogether.

The second effect of the fence is on the local suction peaks near the leading edge. If the wing without fence

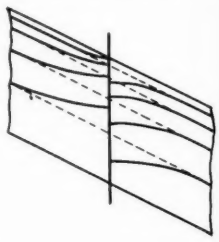

FIGURE 24. Isobars on a sheared wing with plate.

is such as to produce a breakdown of a short bubble near the leading edge at high incidences, with a type of flow as in case *B* in Section 3.4, the fence will increase the danger of this breakdown occurring in its inboard junction. If the flow breaks down there, a part-span vortex sheet will be formed slightly farther inboard which, owing to its curved shape, will soon join the fence itself. This will limit the region of breakdown to a small triangle which will have little significance within a fair range of incidence above that of the initial breakdown. At a later stage, the vortex sheet will move inboard into the wing root. But again, the stalled flow need not " jump " the fence, and the outer wing may be unstalled. This will have a beneficial effect on the pitching moment characteristics. If the outer wing

stalls too, another part-span vortex sheet will be formed there, accompanied by the usual detrimental effects on the pitching moment curve. But the onset of this type of flow should be delayed to a higher overall lift because of the change of the span-loading curve. Moreover, in moving inwards, this vortex sheet will be *held up* by the reduced suction peak outboard of the fence and its movements will be arrested there until, at a still higher incidence, the whole wing stalls.

The first use of fences appears to have been made by P. Jordan[12], 1939, with a view to establishing a fixed boundary between one part of a wing with attached flow and another part where the flow had broken down. This investigation was concerned with transverse fluctuations of the flow in the boundary between the two flow regimes, which have been observed on unswept wings. The fence prevented any sideways movement of boundary layer material and, at the same time, was able to sustain the necessary side force. Such transverse fluctuations of the boundary between two flow regimes may occur again on swept wings. Whether this has any detrimental consequences is not yet known.

Thus the main possible effects of a fence are fourfold: the outward movement of boundary layer material is obstructed; the span-loading is made more uniform; the development of the main part-span vortex sheet is delayed and its movement slowed down in a desirable manner; the downwash at the tailplane is affected.

5. Experimental Methods of Investigating Various Flow Patterns

Because of the practical importance of these mixed flow regimes, some effort will have to be spent on investigating them. The experimental investigation of the vortex patterns is complicated by the fact that not only is some knowledge required about their origin and their characteristics near the wing, but also, the shape of the wake in the Trefftz plane is needed for further theoretical analysis. By this is meant the wake as it would be at a large distance behind the wing if the vortex sheet were carried away by, and in the direction of, the free stream after it has left the wing. Obviously, this does not really take place; the vortex sheet deforms under its own influence, *i.e.* it moves mainly downwards and rolls up into discrete vortex filaments; further, turbulent mixing soon removes any discontinuities in the velocity. It may be expected, however, that, as in ordinary aerofoil theory, any such deformation or turbulent mixing does not greatly affect the flow over the wing and that this can be related with sufficient accuracy to the wake in the Trefftz plane.

Even if a detailed exploration of the flow in the wake had been made, from that would have to be deduced the fictitious wake in the Trefftz plane. We thus find ourselves in the same position as someone who tries to deduce Prandtl's aerofoil theory from the classical investigation of the air flow pattern in the wake of an aerofoil of finite span by Fage and Simmons, 1925, and in particular, to prove that the flow over the aerofoil is substantially consistent with a wake comprising simply a plane vortex sheet of strength $\partial C_L c/\partial y$.

It appears necessary, however, that some detailed wake traverses should be made to find the velocity and vorticity components there. Also, the flow past the wing must be explored in more detail, not only by pressure measurements on the surface but also by measuring flow direction and velocity away from the surface.

Some attempts have been made to visualise the flow by means of tufts, smoke, or liquids on the surface of the wing, such as oil or lampblack suspended in paraffin. The lampblack method, in particular, is conveniently simple. Some early tests have been made by Abbot and Sherman[14] and the method has been taken up again by Black[15] and is in frequent use nowadays. The analysis of such patterns, however, requires special care because the density and viscosity of the oil film are different from those of the air, and the path of an oil particle may thus differ considerably from the path of an air particle as described in Section 4.2. The path of an oil particle is such as to maintain equilibrium between its momentum and the viscous and pressure forces acting on it; the momentum and shear stress may be quite different from those of an air particle in the same circumstances. For example, if the oil used is extremely viscous as compared with air, the oil particles will acquire hardly any speed, but will follow mainly the pressure gradient. If this oil is put on a two-dimensional unswept aerofoil at high lift, the oil on the rear part of the section will move slowly upstream in streaks into the low-pressure region and, in all probability, a large amount of oil will collect near the suction peak or in the short air bubble, if there is one. This mass of oil may even begin to rotate inside the bubble, thus conveying the false impression of a "bubble vortex," *i.e.* the existence of a large circulation within the bubble whereas in fact the total vorticity within the bubble might be trivial. Moreover, if in the course of time more oil collects than the bubble can hold, the bubble may burst, possibly at one or several places, and a comparatively thick streak of oil may be taken along with the air. Again, the false impression of a "vortex pattern in the boundary layer flow" may be given, even on a two-dimensional aerofoil.

This movement of a heavy, viscous oil film is still more complicated on a swept wing with its spanwise flow in the boundary layer and vortex patterns. Consider the example of a wing with a part-span vortex sheet which begins some way out from the wing root, Fig. 25. A short bubble may exist inboard of the vortex sheet, in which some of the oil will collect. With a flow as in Section 4.4, there may be a long bubble outboard of the vortex sheet and the oil is thus less likely to collect there. It is possible that the air (or the oil) is not completely at rest inside the inboard bubble. Since inboard of the vortex sheet, the local lift and the suction peaks near the leading edge normally increase towards the vortex sheet (*see* Fig. 17), the bubble may shorten and the pressure inside it will then fall. The oil in it will thus tend to move spanwise towards the vortex sheet. The bubble is not likely to extend into the wing

root and some of the oil there, after an initial slow movement downstream, will turn round and also flow into the bubble near the vortex sheet from behind. Other oil particles which start farther downstream will not be able to reach the bubble and hence follow the usual spanwise flow pattern. Thus the streaks of oil are likely to form a characteristic pattern, as indicated in Fig. 25, with a relatively big blob of oil near the vortex sheet, and this pattern may convey the idea of a "standing vortex" there, whereas there is certainly no vortex in the usual sense. However, if interpreted correctly, such a pattern may in most cases be taken as an indication that a part-span vortex sheet exists just outboard of the big blob of oil.

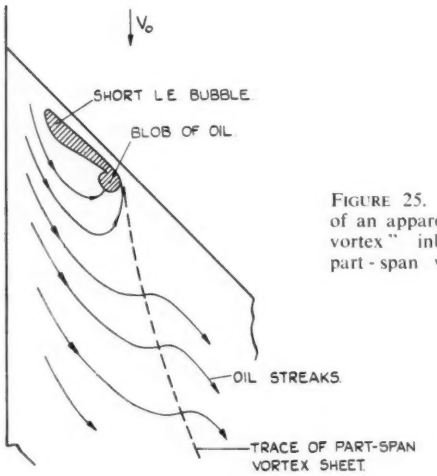

FIGURE 25. Formation of an apparent "standing vortex" inboard of a part-span vortex sheet.

Another indication of the presence of the vortex sheet is a characteristic change of direction of the oil streaks when passing through the sheet, Fig. 25. The oil at the rear of the section will anyway flow outwards and, if the viscosity of the oil is large, the outflow angle must be higher than that of the boundary layer air. In passing through or underneath the vortex sheet in the boundary layer, the oil is affected by an induced velocity component normal to the vortex sheet, which is naturally strongest near the vortex sheet. This means that the oil streaks will turn still farther away from the free-stream direction. This induced velocity will decrease farther outboard on the wing, and the characteristic wave in the oil streaks indicates the position of the part-span vortex sheet. If there is a completely stalled flow outboard of the part-span vortex sheet, this oil will tend to move forward in this region, following the pressure gradient towards the nose of the aerofoil (*see* Fig. 5, case *B*). In that case, the oil streaks do not curve back as drawn in Fig. 25, but continue to turn round against the free-stream direction. In some cases, the part-span vortex sheet appears to detach itself from the surface within the wing chord; the region in which the oil streaks are nearly normal to the free-stream direction is much wider than drawn in Fig. 25.

In other cases where the oil film is relatively thick, a "herring-bone" pattern may emerge. Inboard the vortex sheets the oil may be caught and straightened out by the high-velocity air, part of which has come over the top of the sheets; whereas outside the vortex sheets, slower air or separated flow will allow the oil to follow more closely a spanwise direction. The oil streaks then give the effect of a "watershed," which can be taken as indicating the trace of a vortex sheet.

Tufts on the wing surface normally give a less definite pattern than the oil or lampblack streaks, and they are also subject to some distortions, in particular if they are at all stiff. Tufts attached to a point above the wing surface will always move on a cone if the point of attachment happens to be in, or near, a vortex sheet. This will be seen from the simple case sketched in Fig. 26. Again, the cone formed by a tuft or by a series of tufts, should not be mistaken for the shape of a "vortex."

Streamlines may be indicated by smoke suspended in the air. For a plane vortex sheet, as in Fig. 26, the streamlines will take the shape of spirals. Since a part-span vortex sheet is of semi-infinite length, the circulatory velocity components induced by it increase in a streamwise direction and the spiral formed by smoke widens downstream, *i.e.* it lies on a cone. This may again give the impression that the part-span vortex sheet consisted of a discrete vortex filament in the shape of a large spiral springing from the wing somewhere near the leading edge, as is sometimes supposed in an attempt to "materialise" these vortices.

Since so very little is known as yet about the various vortex patterns, experiments by any of the methods mentioned are desirable and can give useful results, provided that they are interpreted carefully. Tests in a water tunnel can be particularly valuable in this respect (*see* for example H. Werlé [18]).

6. Effects of Reynolds Number and Mach Number

The Reynolds number cannot affect the general character of the various vortex patterns already described but it will greatly influence the changeover from one type of flow to the other, as well as details such as shape and stability of the vortex sheets. The formation of all these vortex sheets (including the ordinary trailing vortices) is associated in some way with a separation of the flow, and the conditions under which such a separation occurs are, of course, affected by Reynolds number. Thus the turbulent separation from the rear of the section, the laminar separation near the leading edge which may lead to the formation of bubbles, and the separation of the flow round the wing tip, all depend on Reynolds number. Whether a short or a long bubble is formed near the leading edge also depends on Reynolds number, as suggested by P. R. Owen. Little detailed knowledge is available as yet.

There is no reason why the principal features of the vortex pattern should change with Mach number up to the critical speed. Local supersonic regions and the presence of shock waves at higher Mach numbers, however, must have a profound effect on the actual development of the separated regions. For example, two

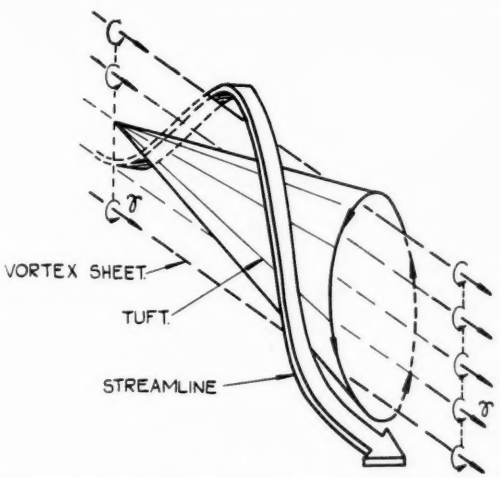

FIGURE 26. Conical motion of a tuft attached to a point in a vortex sheet and streamline.

aerofoil sections with different nose thicknesses will have different suction peaks at low Mach numbers and hence bubbles will be formed at different C_L values; even different types of bubble may occur. If the flow becomes supersonic locally (and this may occur at fairly low free-stream Mach numbers) the local Mach number will not exceed a certain limiting value, 1·3 say for aerofoils of moderate thickness, or even infinity (vacuum). Hence, at a certain free-stream Mach number, both aerofoils may have arrived at the same limiting Mach number, in which case the difference in suction peaks will have disappeared and with them possible differences in flow types. It is thus possible that two aerofoil sections which have for example, different $C_{L\,max}$ values at low Mach numbers arrive at the same $C_{L\,max}$ values at a higher

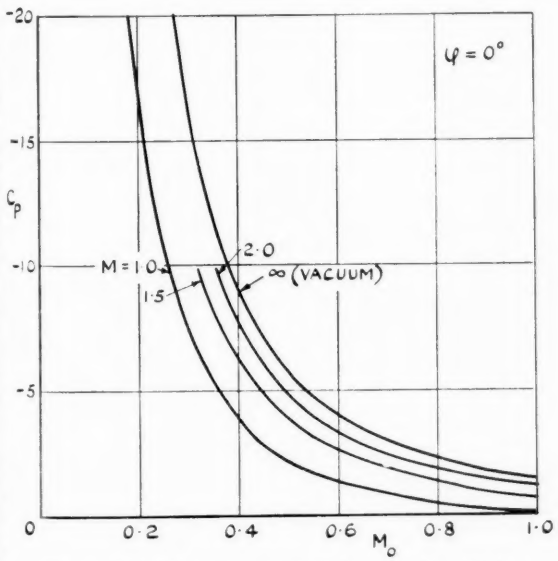

FIGURE 27. The pressure coefficient to give velocity of sound, or vacuum.

Mach number. To illustrate this, lines of constant local Mach number, M, corresponding to certain pressure coefficients, C_p, and free-stream Mach numbers, M_o, have been drawn in Fig. 27. Whereas, at low M_o, short bubbles are associated with C_p values of the order of -10, such high suctions cannot possibly occur at high Mach numbers. The mechanism which leads to a breakdown of the flow and to the formation of bubbles at high Mach numbers has yet to be investigated.

The presence of a supersonic region will also affect the formation and shape of the bubble, as well as its stability. In other cases, a shock may itself be the cause of rear separation. In the present context, it is of interest to note that it is doubtful in such a case whether the separated regions will be divided from the non-separated regions by a single free boundary, or whether the transition is more gradual; this would complicate matters considerably because it might necessitate the introduction of substantial volumes in which the vorticity is not zero.

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Developing Aviation Fuels and Lubricants*

by

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Introduction

The purpose of this paper is to illustrate the research and development work necessary to improve existing products and to anticipate demands for new products facilitating the advance in efficiency and performance of aircraft. The experience of Thornton Aero Engine Research Laboratory, in which the author has worked since it came into operation in 1941, has been drawn on almost entirely.

Aviation Fuel

LEADED GASOLINE

The use of tetra-ethyl lead (T.E.L.) as an additive to gasoline, to increase the knock-limited power available from the spark ignition reciprocating engine, has made a major contribution to the development of aviation. The inclusion of progressively increasing concentrations of T.E.L. has been necessary in the provision of high grade gasoline in the quantities required for worldwide military and civil flying in peace and war. It has therefore been necessary to overcome the engine problems which arise due to the presence of ethyl fluid. These problems increase in severity with increasing concentration of the additive.

During the 1939 to 1945 War, unlike the Germans, British and American engines employed carburettors to provide fuel/air mixtures. The T.E.L., the bromine compound used with it as a scavenger for lead combustion residues, and the fuel itself possessed different volatility characteristics. Particularly if induction manifold temperatures were low, when each individual cylinder of a multi-cylinder engine received its charge the distribution of these fuel components did not remain the same as in the original fuel. Some cylinders received too little lead and tended to detonate; others received too much, which promoted combustion chamber problems such as exhaust valve and spark plug deterioration.

Figure 1 shows the type of sampling equipment necessary for measuring lead distribution, and Table I illustrates the distribution pattern found in an engine.

Several ways in which this maldistribution could be overcome were recommended but, as engine outputs have increased and weaker mixtures have been used, some of the problems familiar from this work have grown in importance, particularly that of spark plug fouling. Combustion chamber studies have been extended, therefore, to find ways of improving the useful life of spark plugs. In airline service these components may have a life of about 500 hours, cleaning being necessary after each 100-200 hours. The necessity for cleaning, and many premature plug failures, result from the lead compounds which do not leave the combustion chamber as they are intended to. The bromine compound is included in ethyl fluid in just the right proportion to convert all the tetra-ethyl lead to lead bromide during combustion, because this compound has a much higher volatility than the lead oxide and sulphate which otherwise would result. Nevertheless, investigation has shown that not all the lead, or even all the bromine, leaves the combustion chamber according to this idealised chemistry. In fact, as shown in Table II, a number of otherwise unfamiliar lead compounds have been identified by X-ray diffraction analysis of the small quantities of deposit which can be isolated in various areas of the combustion chamber.

The melting point and the electrical conductivity of individual members of this complex range have been determined, and Fig. 2 contains some of this information. Correlating this information with the appearance of failed spark plugs and analysis of spark plug deposits showed that, in many cases, plug failure was due to semi-conducting coatings of lead oxy-bromides impairing the efficiency of the insulator. Some experiments were conducted in a simple continuous combustion rig to evaluate additives which might raise the melting point of the lead residues and reduce their electrical conductivity. It was found that an organic phosphorus compound, added to the fuel in extremely small amounts, had the best all-round prospects for this application. The rig tests were then superseded by a full investigation, using single-cylinder aero-engines in the laboratory and subsequently confirming the good results in multi-cylinder engines and flight trials.

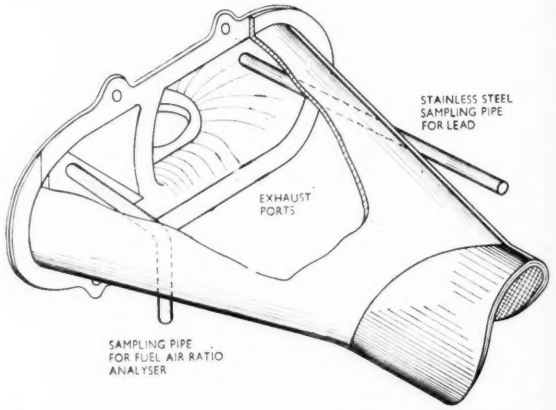


FIGURE 1. Arrangement of gas sampling points for measuring lead distribution.

**A Lecture read before the Bristol Branch of the Society on 25th November 1952 and to the Gloucester and Cheltenham Branch in December 1952.*

TABLE I

DISTRIBUTION OF T.E.L. IN A WARTIME U.S. V-12 CYLINDER LIQUID-COOLED ENGINE

Fuel:—100/130 grade; 7·2 ml. T.E.L./I.G.
Air intake temperature:—20° C.

Cylinder Bank	*T.E.L. ml./I.G. to nearest ml. Cylinder number*					
	1	2	3	4	5	6
Left	5	8	6	6	6	6
Right	13	10	9	6	6	7

Tri-cresyl phosphate is a stable, colourless, syrupy, non-toxic liquid of about the same volatility as T.E.L. It is readily miscible with gasoline, and its use could, for example, allow spark plugs in airline service to operate in excess of their normal lifetime of about 500 hours, with removal dictated by electrode erosion rather than necessity for cleaning.

TURBINE FUEL

In contrast with the chemically specialised fuels required by the high specific output reciprocating engine, it has been shown that the gas turbine engine is not as selective in its fuel requirements.

Apart from the important consideration of carbon deposition, the chemical nature of a turbine fuel is unimportant, except under extremes of operating conditions on which research is still in progress. Physical properties, however, are important. For example, a high volatility may facilitate relighting at altitude. Design action can overcome many of the differences in performance due to variation in physical properties of fuel. For military purposes the specification of a turbine fuel is therefore governed, within wide limits, by the extent to which the material could be made available in emergency. For civil use, the future should see the best compromise effected between adequately low freezing point and a low volatility, to promote safety in the event of mishaps during aircraft operation.

The sensitivity of turbine fuel systems to small particles of foreign matter makes it essential to subject fuel to microfiltration during delivery to an aircraft.

When contaminating solids have been completely removed during the supply of the fuel to the aircraft,

TABLE II

COMPOSITION OF COMBUSTION CHAMBER DEPOSITS
Fuel:—Aviation fuels of various lead contents

	Chemical formulae of some of the lead components identified			
Piston crown	$PbSO_4$	$2PbO.PbBr_2$		
Cylinder head	$2PbO.PbBr_2$	$2PbO.3PbBr_2$		
Inlet valve	$2PbO.PbBr_2$			
Exhaust valve	$3PbO.PbBr_2$	$4PbO.PbSO_4$		
Spark plug Insulator	$2PbO.PbBr_2$	$PbO.PbBr_2$	$PbO.PbSO_4$	$2PBO.3PbBr_2$

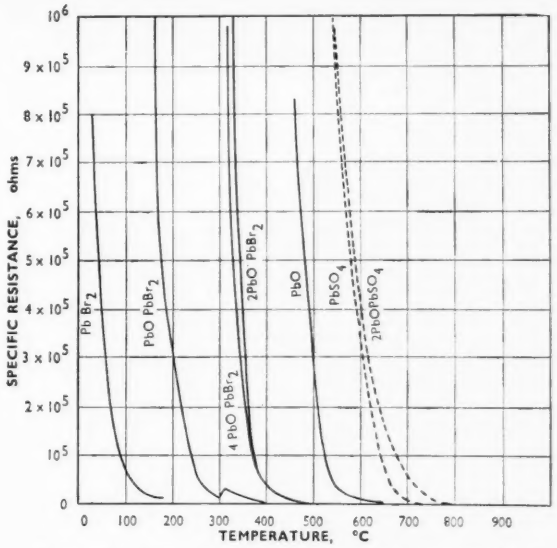


FIGURE 2. Comparative resistivities of lead compounds.

it is still necessary to consider what precipitation of solids may still arise in the fuel when low temperatures are experienced in flight. The freezing point of the fuel is controlled by specification. The freezing point represents the temperature at which the first separation of wax-like hydrocarbons occurs in the fuel. These waxes would cause blocking of filters in the aircraft fuel system, but no possibility of their separating arises above the freezing point of the fuel.

Unfortunately, some water dissolves in kerosines and gasolines, and, as shown in Fig. 3, the solubility of this water decreases as the temperature of the system is reduced. It is not possible to keep a fuel absolutely dry, and the relatively small solubility of water in the fuel leads to saturation concentrations of water readily occurring in kerosine at normal atmospheric temperatures. Precipitation of some water is therefore inevitable during flight under conditions involving low fuel temperatures. The precipitated water droplets will freeze at temperatures well above the freezing point of the fuel itself. Fuel freezing point specifications do not, therefore, protect against the hazard of filter clogging by ice particles. The ice particles may be small enough to pass through normal filters, but in some cases, particles can agglomerate and build up in filter material until clogging is achieved. The precise differences in conditions governing these two possibilities have not been thoroughly elucidated, but methods for dealing with the potential problem of filter icing are necessary.

There are several ways of dealing with ice formation: the filter or the fuel may be heated or the fuel may be doped with a small quantity of methanol. This latter ensures that any water separating from the fuel will contain enough methanol to reduce its freezing point to below that of the fuel.

In addition to liquid anti-freezes of the methanol type, it is well-known that solids can behave as anti-freeze materials, for example common salt is known to

depress the freezing point of water. The possibility was investigated of using a canister of solid anti-freeze in the fuel system of an aircraft, so that water separating from the fuel would pick up enough of the solid to keep its freezing point below the specification figure for the fuel.

Figure 4 shows the efficiency of various solids in reducing the freezing point of water and it will be seen that chromium tri-oxide is remarkably effective in this respect. Rig tests showed that a canister of chromium tri-oxide could treat water-saturated fuel passing through it so that ice formation, caused by water separating at low temperatures, was suppressed well below – 40°C. The use of such a canister in a fuel tank might mean that water saturated with chromium tri-oxide could collect at the bottom of the tank, so corrosion tests were made with various metals of importance in tank construction. Since oxidising agents, such as chromates, are well-known inhibitors for the corrosion of metals such as iron and aluminium, on which they encourage protective oxide film growth, it was not surprising to find that the chromium tri-oxide solution caused no corrosion. Unfortunately, however, this aqueous oxidising solution was destructive towards non-metallic material, such as rubber-like sealing compounds used in the construction of integral wing fuel tanks and therefore, a canister in the tank would not be an attractive proposition. It would still appear to be practicable to include a canister in the fuel line from the tank to the filter, provided that there was no chance of a pool of aqueous oxidising agent collecting in contact with any non-metallic seal or flexible pipe. However, recent information has suggested that silver-plated components in fuel systems might be corroded by kerosine containing chromium tri-oxide, so that the advisability of any specific application could only be decided after further rig work using a complete range of samples of the appropriate constructional materials.

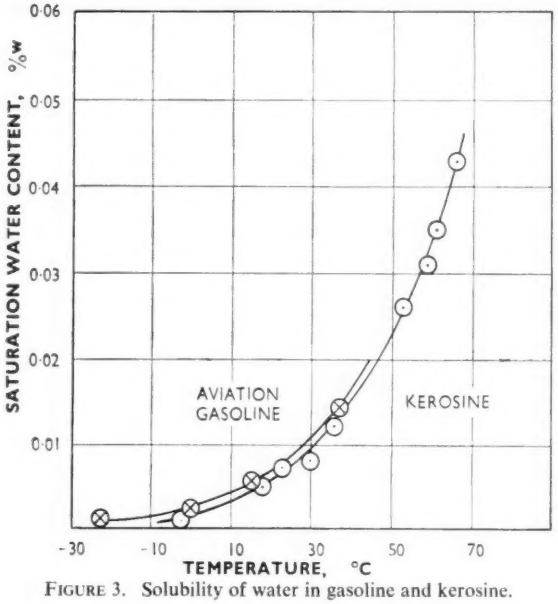


FIGURE 3. Solubility of water in gasoline and kerosine.

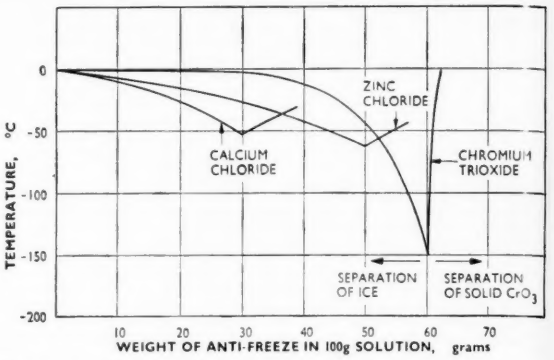


FIGURE 4. Solubility/freezing-point data for solutions of some solid anti-freeze materials in water.

Lubricating Oils

The aviation reciprocating engine is adequately served by current straight mineral oils which are of relatively high viscosity and viscosity index, and provide satisfactory film strength for the lubrication of reciprocating contacts, gears and bearings at high temperatures. For operation in cold climates, there is an effective method of dilution of the oil with fuel before shutting down an engine. After the cold start with the diluted oil, the diluent evaporates rapidly as the engine reaches working temperature. The high film strength of the undiluted lubricant is thereby re-established as conditions approach normal severity in the engine. In the gas turbine engine, the nature of the fuel and of the oil system precludes the use of a dilution procedure to promote pumpability of the oil at low temperatures. The oil must itself be able to meet requirements at the lowest temperatures which will be encountered. The ability to do this will depend on the viscosity-temperature characteristics of the oil, but pumping experiments with an actual gas turbine oil system were necessary to determine precisely at what value viscosity became a limiting factor in the performance of the system.

A turbo-prop system was selected for tests in view of its greater complexity, which would therefore impose more severe limitations on oil viscosity than the turbo-jet system. Figure 5 shows the oil system installed in the Thornton Research Centre Cold Room for test and Fig. 6 some oil pump performance curves with specification mineral oils at present in use in turbine aircraft, namely, D.Eng.R.D.2479 and 2480, DEF.2001, and the reciprocating engine oil D.Eng.R.D.2472B.

The important conclusions from this work were that, while more or less complete cessation of oil flow in the system did not occur until the temperature had fallen to the pour point of the oil, full volumetric output of the pump could not be maintained when the temperature reached a level at which the viscosity of the oil entered the 3,500-5,000 centistokes range. An exact viscosity is not quoted because the figure would depend on pump speed, as shown in Fig. 7. The higher the pump speed, the lower the viscosity at which full output was first impaired.

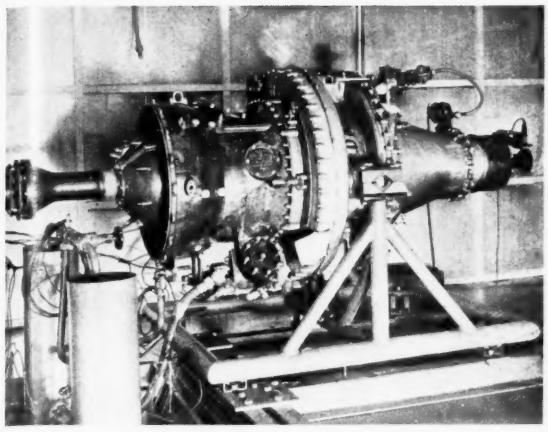


FIGURE 5. Turbo-prop oil system in cold room.

Having completed these calibration experiments, oils can be selected from the point of view of low temperature performance by inspection of viscosity-temperature plots. Some plots for specification mineral oils are shown in Fig. 8.

$-40°$ can be taken as an oil temperature below which it should not be necessary to cater for full volumetric output in conventional aircraft.

The conclusion follows that DEF.2001 oils would be almost acceptable lubricants from this point of view. Unfortunately, the oil in a gas turbine has also to perform some exacting lubrication duties at high temperatures and speeds, in respect of which DEF.2001 oils have deficiencies. Taking the simplest deficiency, DEF.2001 oils are relatively volatile, as shown in comparison with other specification oils in Table III. This leads to undesirably high oil consumption figures in any type of gas turbine.

Again considering turbo-jet and turbo-prop designs, the lubrication of high-speed ball bearings makes severe demands on the oil. Rolling does not involve full fluid oil films at the contact, but when the rolling components are not markedly deformed, coefficients of friction are low, of the order of 0·001 with, or without, oil present.

TABLE III

VAPOUR PRESSURES OF DEF.2001, D.ENG.R.D.2479 AND 2480 OILS

Temperature	*Vapour pressures*, mm Hg		
C	2001	2480	2479
	($\times 10^{-3}$)	($\times 10^{-4}$)	($\times 10^{-4}$)
20	0·42	0·167	0·062
30	1·02	0·47	0·163
40	2·35	1·24	0·41
50	5·4	3·08	0·95
60	10·6	7·3	2·12
70	21·3	16·2	4·30
80	40·5	34·7	9·2
90	74·8	71·1	18·2
100	134	140	34·0
110	232	267	63
120	390	490	110
130	642	861	193

Finite deformation, even if small, always takes place on rolling, and the degree of deformation is related to the rolling friction. With the very small deformations experienced in ball and roller bearings, it is at first sight difficult to see what part a lubricant will play other than taking heat away from the unit, protecting against corrosion, and lubricating incidental sliding contacts such as those occurring with ball cages. However, it must be remembered that high-speed, heavily-loaded ball races will eventually fail by fatigue of balls and track through repeated application and removal of the deforming stresses. The oil is able to take a part at the site of metal deformation and reduce the severity of the stress reversal schedule.

It is usually considered that a sliding component, introduced due to deformation at a rolling contact, largely accounts for the so-called rolling friction. Tabor[1] has recently challenged this view and has suggested that the appropriate areas enter and leave a rolling contact in a deformed state so that no slip actually takes place. Tabor attributes rolling resistance to the hysteresis loss in the energy cycle involved during the repeated elastic compression and expansion of the materials entering and leaving a rolling contact. In this case a lubricant could prolong the life of rolling elements by the removal of heat and possibly, by distributing deformation over a larger area with correspondingly reduced intensity at any site within the area.

An attempt has been made to measure the influence of different lubricants on the life of a heavily-loaded, high-speed bearing. This task has proved extremely difficult because of the poor repeatability of bearing life

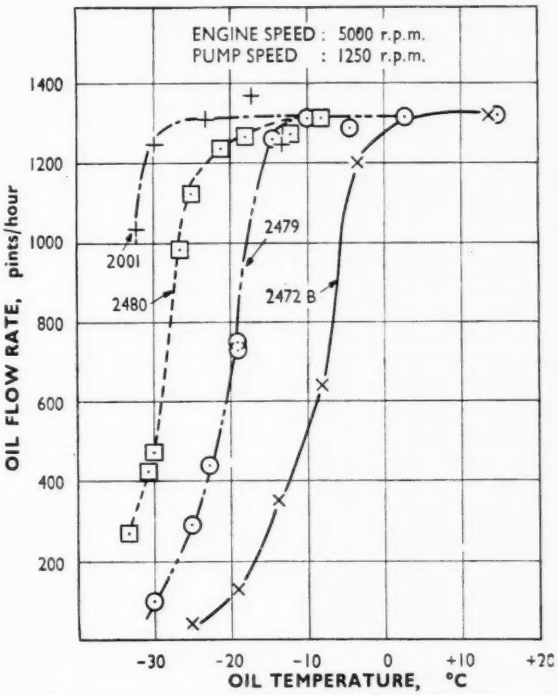


FIGURE 6. Variation of oil flow rate with temperature.

experienced with one oil and a succession of nominally identical test bearings, but statistically significant improvements in bearing life were established when a low viscosity oil, such as DEF.2001, was replaced by a high viscosity oil, such as D.Eng.R.D.2472B. The influence of more moderate changes in oil viscosity could not be established because of the "scatter" of results.

In the turbine engine the oil is required to lubricate high-speed gears. This problem has been investigated using power circulating rigs capable of measuring the load at which gear scuffing occurs under a given set of conditions. A $3\frac{1}{4}$ in. centres rig has been used up to speeds of 6,000 r.p.m., and a 5 in. rig up to speeds of 15,000 r.p.m. Figs. 9 and 10 show that, for any given oil, scuffing load decreases with increasing speed and, for a series of high (H.V.I.) and medium (M.V.I.) viscosity index mineral oils, the higher the viscosity the higher the load-carrying capacity.

Summarising the consideration of gas turbine lubricants, it must be emphasised that seriously conflicting requirements occur at opposite ends of the very wide operating temperature range. An oil is required which has the viscosity of DEF.2001 oil at low temperatures and approaches the viscosity of D.Eng.R.D.2472B oil at high temperatures. This represents a viscosity index of at least 150, which it is impossible to achieve with mineral oils.

The use of additives in mineral oils was then considered. A small quantity of a so-called "Extreme Pressure" additive will raise the load-carrying capacity in gears of a low viscosity oil to a figure appropriate to a higher viscosity undoped oil (Fig. 11). "Extreme

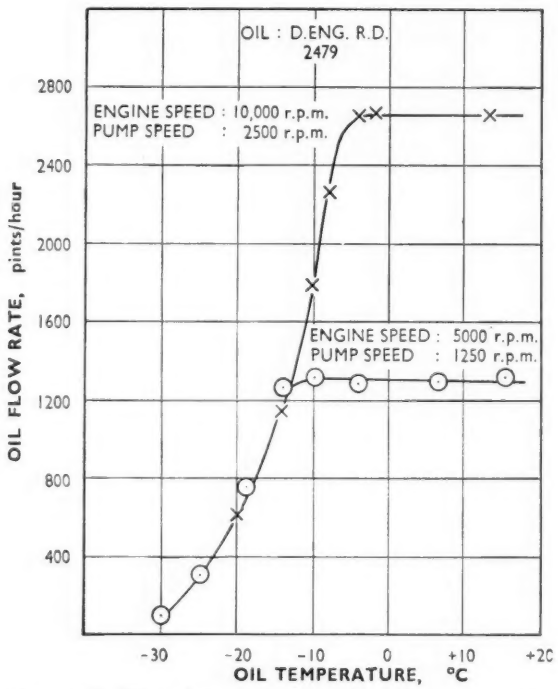


FIGURE 7. Effect of pump speed at various oil temperatures.

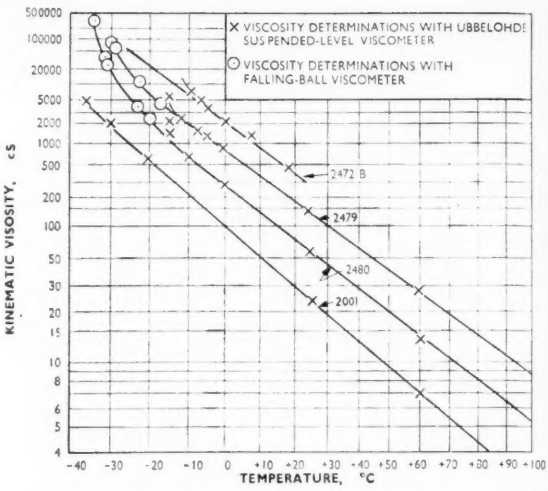


FIGURE 8. Viscosity-temperature diagram based on A.S.T.M. standard viscosity-temperature chart.

Pressure" lubrication deals with very severe sliding conditions under which full fluid film lubrication is impossible, and the key to the severity is the high temperature developed in the contact. The conditions immediately preceding scuffing in a gear test with a straight mineral oil are on the threshold of the extreme pressure range. The oil can be doped with an additive which has been selected to be inert under the less severe conditions of lubrication, but capable of forming a certain type of chemical film on a gear tooth surface at localised points where temperatures rise rapidly as a prelude to the complete lubrication failure which would have occurred in the absence of additive. The extreme pressure film is formed by chemical reaction of the additive with the metal; the reaction is triggered and sharply accelerated by the rising temperature condition of incipient localised seizures on the metal surface. It appears that the actual layers of reaction product form an emergency lubricant preventing metal-to-metal welding, which would otherwise occur. It can be imagined that any fairly refractory material would interfere with the metal-to-metal welding, just as do foreign bodies in conventional welding, until fluxed away. Very special characteristics are considered necessary for the real success of an extreme pressure film. A merit factor can be constructed for such films:

$$\text{Merit Factor} = \frac{T}{\mu}$$

where T is the temperature at which the film itself melts to a low viscosity liquid affording no further protection, and μ is the internal or shearing friction of the film below T.

Accordingly, high merit is achieved when high melting temperature is combined with low shear strength below that temperature. Easily sheared inorganic compounds, such as iron chloride, have the necessary properties to be effective as extreme pressure layers. It is quite practicable to use easily sheared

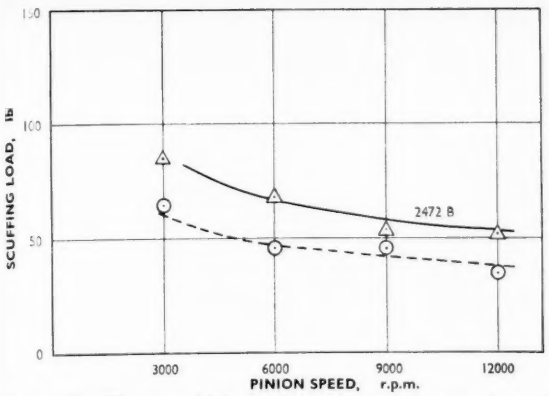


FIGURE 9. Thornton high-speed gear rig. Influence of speed on scuffing load (24/26 ratio gears).

chlorides as extreme pressure films through the addition of a suitable chlorinated hydrocarbon to a mineral oil.

Although it would be possible to get over the gear load-carrying capacity deficiencies of low viscosity oils in this way, it is clear that volatility, affecting oil consumption, would not be reduced. Also, the cushioning effect of high viscosity in prolonging the life of high-speed ball bearings would not be achieved by a doped low viscosity oil. For the future, therefore, a demand arises for a fluid which has viscosity-temperature characteristics superior to those attainable with a mineral oil. There are no naturally available fluids with such properties, but they can be achieved in several synthetic fluids, although it must be realised that, when synthetic materials must be resorted to as lubricants, the cost goes up greatly relative to that of mineral or vegetable products. The ester oils—fluids containing carbon, hydrogen and oxygen, and selected as the base material for the formulation of a superior aviation turbine oil—will cost £3-£4 per gallon, and can only be obtained at that figure because similar materials have application to the extent of about 40,000 tons a year as plasticisers in the growing plastics industry. The aviation demand will not exceed a few thousand tons a year for several years.

Ester oils, suitably blended with other components, result in products which at high temperatures have low volatility, high load-carrying capacity, high viscosity and thermal oxidation stability, while at low temperatures, through their viscosity index (which is over 150) they can be pumped to a temperature of $-50°C$. The load-carrying capacity of such oils is even higher than would be expected from viscosity considerations in comparison with mineral oils. However, ester oils are not entirely free from disadvantages. Perhaps because of their good lubricating properties, they creep through seals and small clearances more than do mineral oils, and have an adverse effect on many sealing, insulating and paint materials developed over the years to have satisfactory resistance to mineral oils. The Ministry of Supply issued a tentative specification—D.Eng.R.D.2487—for such oils in March 1952 but, in view of the satisfactory service which high grade mineral oils are giving in the de Havilland Comet, for example, it is clear that it will be some time before civil aviation will need to take advantage of the extreme range of properties which can be realised, at a price, with synthetic oils.

Speciality Products

In addition to its main requirements of fuel and lubricating oil, an aircraft needs a range of special greases and fluids. These products must also perform satisfactorily over a wide range of temperatures.

A grease usually consists of a mineral oil thickened and given a structure by means of soap fibres which form a separate phase acting as a sponge for the oil. As for turbine oils, the use of ester oils instead of mineral oils extends the temperature range in which the soap-type grease can operate. A further extension is possible at the high temperature end by using an inorganic microgel, instead of a soap, to give the synthetic oil a grease structure. The microgel consists of a loosely-packed aggregate of roughly spherical particles well below one micron in diameter, whereas a soap matrix may consist of soap fibres several microns long and 1/10 or 2/10ths of a micron in diameter.

Recently, it has become possible to make a microgel grease which will lubricate a sealed actuator required to work within the temperature range -60 to $+200°C$. Future developments may permit a further extension of the practicable operating range at the high and low temperature ends of the scale by using other synthetic oils in the microgel matrix.

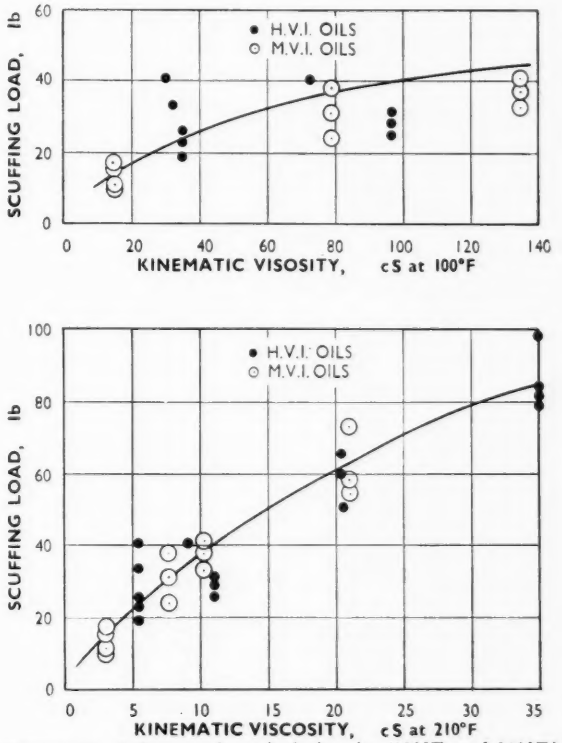


FIGURE 10. Influence of nominal viscosity (100°F. and 210°F.) on scuffing load.

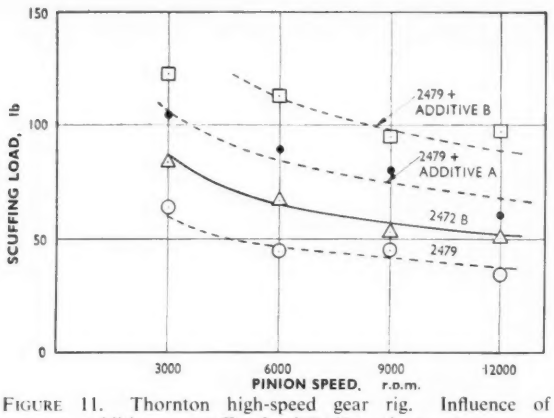


FIGURE 11. Thornton high-speed gear rig. Influence of additives on scuffing load (24/26 ratio gears).

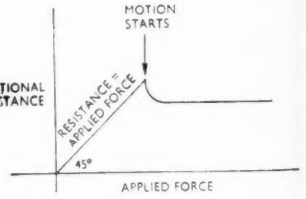
Lubrication problems are more likely to arise with grease-lubricated equipment than for items with oil lubrication. For example, fretting corrosion can occur on greased rubbing surfaces where persistent small oscillations under load eventually push all the lubricant away from the vital areas of contact. The amplitude of the oscillations is too small to involve replenishment of the contacts with grease during any temporary relaxation of the load. All the aspects of fretting corrosion have not been elucidated, but it appears that, first of all, the highspots actually in contact at a site of fretting rid themselves entirely of even boundary lubricating films. This raises the friction at these points and appreciable areas of the contact reach temperatures at which chemical reaction of the metal occurs with the oxygen of the atmosphere. This produces metal oxides which may be abrasive, as is the case with iron oxide, and the presence of such material increases the localised damage by lapping. However, even without the formation of oxide, damage would occur because of the large-scale formation and subsequent disruption of the innumerable microwelds which are considered to account for friction between unlubricated surfaces. In this case, metal particles would be torn from the surfaces and could subsequently add to the damage by a process of abrasion. This point has been emphasised to show that the exclusion of oxygen does not necessarily prevent the occurrence of fretting. Reduction, or even elimination, of the damage to chafing components can be achieved by preventing the complete disappearance of lubricant from any site in the contact. One way of doing this is to include a relatively high concentration of an active "Extreme Pressure" additive in the grease. As the grease film thins between chafing surfaces and localised hot-spots develop at the asperities in the contact, chemical reaction between the extreme pressure agent and the metal surfaces will occur, just as described in detail when gear scuffing was considered. The extreme pressure film will provide emergency boundary lubrication, and, being attached chemically to the surfaces, it will strongly resist displacement.

In Dr. Bowden's laboratory at Cambridge, it has been demonstrated by the sensitive radio-active tracer technique that, in comparison with unlubricated sliding, a boundary lubricant film reduced the damage of sliding metals by a factor of more than 20,000. Incidentally, this showed up as only a 20-fold reduction in the coefficient of friction, which confirms the fact that the coefficient of friction is an insensitive guide to the damage being suffered by a sliding surface.

The practical significance of these statements is illustrated by the fact that special Extreme Pressure (E.P.) greases, recommended as a result of rig tests at Thornton, have been successful in alleviating fretting corrosion in the Hardy Spicer couplings of auxiliary gearbox drive shafts and in open-ended splines involved in generator, hydraulic and vacuum pump drives.

At this point it is necessary to mention the application of materials such as graphite and molybdenum disulphide as boundary lubricants. These materials possess the high softening point and low shear strength characteristics, in a direction parallel to their pronounced cleavage planes, considered desirable for emergency protection of rubbing metals. However, if these materials are merely dispersed in a grease there can be no guarantee that the suspended agent will arrive and be retained at the critical areas of a severely loaded contact just when required and in the necessary quantity. The chemical E.P. agent, functioning through chemical reaction in place, has a great advantage. To support this idea, it has been found that, when a material such as molybdenum disulphide is bonded on to surfaces by baking a suspension in corn syrup in place, the surfaces acquire excellent resistance to fretting. Similarly, a bonded phosphate coating is effective, and outstanding merits are claimed for such phosphate coatings subsequently impregnated with molybdenum disulphide.

FIGURE 12. Influence of applied force on frictional resistance during sliding without lubricant.



Although not directly connected, the subject of fretting can now be followed by consideration of screw-jack lubrication. Provision for the lubrication of these units may involve grease or a fluid. One of the problems to be avoided is the occurrence of a judder or stick-slip oscillation, instead of smooth operation of the jack under load. In dealing with this problem it is necessary to distinguish between static and dynamic friction under boundary lubrication conditions, *i.e.* in the absence of a full fluid lubricating film. Fig. 12 shows the relationship between frictional resistance and applied force in a system for which the static friction is greater than the dynamic. This is generally the case. If the elasticity of the system through which the force is applied is sufficient to permit a reduction in this force once motion is achieved, the load may come to rest again and remain at rest until the applied force has built up once more to a value which can overcome static

friction. A repetition of this sequence has been termed "stick-slip" by Dr. Bowden.

Really efficient, but not all, boundary lubricants overcome this intermittent motion by altering the slope of the friction-velocity curve, and lubricants of the extreme pressure type can, in certain cases, be chosen to make the coefficient of static friction equal to, or even less than, the dynamic. However, the alternative and usually more attractive method of controlling the severity of the vibrations encountered in stick-slip motion is by attention to the elastic constants of the system, with a view to damping out the fluctuations in applied force.

Having disposed of the problem of stick-slip, it is necessary to decide what factors in the lubrication of a screw-jack will minimise wear of the nut, leading to excessive backlash. This study has recently been started at Thornton, and no conclusive results have been obtained as yet but, from the anti-wear and extreme pressure agents now available, it should be possible to formulate a premium lubricant for this application.

In dealing with speciality products, it is perhaps of interest to mention hydraulic oils. It has been considered that mineral hydraulic oils may often be responsible for initiating the fires which sometimes cause, or so frequently follow, mishaps in flight. Among other things, this idea is based on the fact that, in certain types of spontaneous ignition test, hydrocarbons in the lubricating oil range have appreciably lower ignition temperatures than volatile fractions of the fuel type. In addition, it is emphasised that hydraulic systems involve relatively high pressures and traverse a major proportion of an aircraft's structure which increases the chances of a fractured line spraying oil on to a potential source of ignition. These suppositions have provided the incentive for the development of hydraulic fluids which are considered to be less inflammable than mineral oils. Of these, water-base and phosphate ester types have received most attention. The water-base fluids have shown deficiencies in lubricity, low temperature properties, and residual inflammability following evaporation of water to leave a residue of the "thickener" used in these fluids. Phosphate ester fluids are good lubricants, can have satisfactory low temperature properties and seem an attractive proposition, apart from the disadvantages that a hydraulic system must be sealed and adjacent electrical wiring insulated with otherwise unconventional materials. Nevertheless, an impartial assessment of the inflammability leaves some doubts about the extent of the advance over mineral oils for spontaneous ignition under conditions which might arise in a flying accident.

The spontaneous ignition temperature of a material is one of the most difficult properties to state unambiguously. In a test which measures ignition temperature under quiescent and confined conditions, concentrating primary reaction products around the potential source of ignition, self-ignition temperatures are relatively low. The A.S.T.M. self-ignition temperature test is of this type and the results for hydrocarbons show some relationship to octane number, although detailed study proves that this is not the only

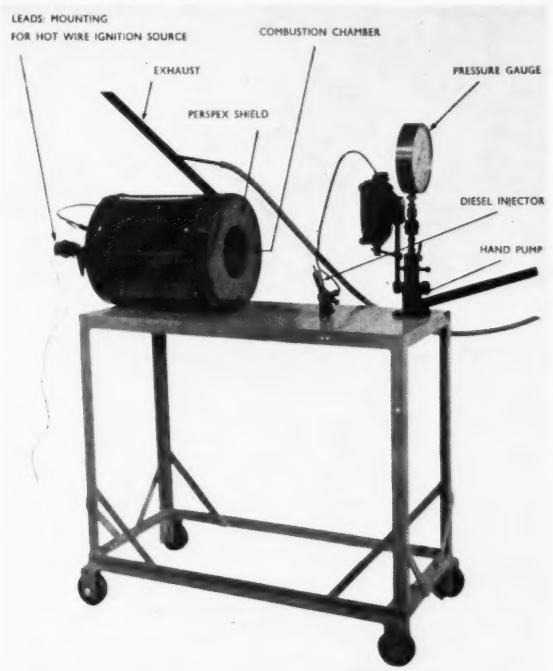


FIGURE 13. Spray ignition test apparatus for measuring the inflammability of hydraulic oils.

property involved. A conical flask is immersed in a bath which allows the bottom of the flask to be held at any selected constant temperature. Drops of test fluid are dropped through the narrow neck on to the base of the flask, at different temperatures, until the minimum temperature for ignition has been defined. Repeatability is good, and this test shows aromatic phosphates to great advantage over mineral oils. Table IV gives results obtained using the A.S.T.M. method. In the same table, some figures for self-ignition temperatures determined by dropping the fluid on to an ordinary hotplate are recorded which differ considerably from the A.S.T.M. figures.

Departing even further from the A.S.T.M. conditions, a spray test (Fig. 13) has been set up at

TABLE IV

SPONTANEOUS IGNITION TEMPERATURES BY A.S.T.M. AND HOTPLATE METHODS

Fluid	*Spontaneous Ignition Temperatures, °C*	
	A.S.T.M.	*Hotplate*
73 octane gasoline	299	
92 octane gasoline	390	
100/130 grade gasoline	429	730
Kerosine	254	650
DTD 585	263	600
DEF.2001	345	430
Tricresyl phosphate	553	
Trioctyl phosphate	380	
Chlorofluorocarbon (CFC)	613	
CFC + polymer to match DTD 585 viscosity	473	

TABLE V

SPONTANEOUS IGNITION BY SPRAY AND A.S.T.M. TEST METHOD

Fluid	*Spontaneous Ignition Temperatures, °C*	
	Spray test	*A.S.T.M. test*
DTD 585	820	263
DEF.2001	820	345
Tricresyl phosphate	840	553
Trioctyl phosphate	720	380
CFC	No ignition at 1,200	613
CFC + polymer to match DTD 585 viscosity	1010	473

Thornton that involves abundant ventilation and turbulence around the source of ignition, which is an electrically heated filament. A small volume of test fluid is projected on to the filament using a hand-operated pump and diesel injector working at about 2,000 lb./in.^{2} pressure. The minimum filament temperature observed for ignition to occur at, or near, the filament was recorded for different fluids. Filaments of different heat capacity, and even a large induction-heated ball, were used as sources of ignition without altering the magnitude of the results appreciably. It was necessary to reach filament temperatures of about 750°C. before ignition of any mineral hydraulic oil occurred. Even aromatic phosphate esters showed no significant improvements on this figure, but a low level of inflammability was achieved in a hydraulic fluid consisting mainly of a chlorofluorocarbon, namely, a hydrocarbon in which about half the hydrogen is replaced by chlorine and the other half by fluorine. Results are illustrated in Table V.

For the time being, the use of halogenated fluids is not considered attractive because of their high density and the results of the spray test leave some doubts about the all-round reality of the advances hoped for in the adoption of some existing so-called non-inflammable hydraulic fluids.

Conclusion

It is emphasised that in all the Thornton investigations of the type referred to, any apparent success in the laboratory must be followed up and proved by participation in the initiation and observation of full scale tests and, eventually, flight tests. The co-operation received from the Aircraft Industry, the Airlines and the Government is valued very highly by the Thornton Research Centre which looks forward to continuing development along these lines.

REFERENCE

1. TABOR, D. (1952). The Mechanism of Rolling Friction. *The Philosophical Magazine*, October 1952.

Fourth Anglo-American Aeronautical Conference

THE FOURTH ANGLO-AMERICAN AERONAUTICAL CONFERENCE, held in London from 15th to 17th September, is generally considered to have been the most successful so far. Judged by numbers alone, it certainly was. In 1951 there were 540 delegates; this year there was a total of 729, of whom 150 were American. The delegates included not only British and Americans, but representatives from many countries in the British Commonwealth and the continent of Europe—all members of the Society or the Institute—in addition to a number of representatives of the Advisory Group for Aeronautical Research and Development of the North Atlantic Treaty Organisation.

Once again many of the American delegates were accompanied by their wives and families, who added greatly to the pleasure of the social functions.

It was originally the intention of the Councils of the Society and the Institute to held this Fourth Conference in the United States; but, as the President of the Society, Sir William Farren, explained in his message of welcome to the delegates—

> "The exceptional circumstances of this great year 1953 obliged the Councils to face the choice between postponing the Fourth Conference until 1954, or holding it in England.
>
> "There was a very strong feeling that to postpone it would be most unwelcome to all who believe, as I do, that these Conferences are invaluable in advancing the profession of Aeronautics. The Council of the Royal Aeronautical Society gladly undertook the task of organising the Fourth Conference in London. . . ."

The Society was fortunate in having the hospitality of University College, London, as the headquarters of the Conference but, as the number of delegates exceeded 500, larger accommodation for lectures had to be found; fortunately, accommodation was found at Friends House, in the Euston Road a few minutes walk from University College. Thus all lecture sessions were held in Friends House while lunch, morning coffee and afternoon tea were available each day at University College.

The courtesy and co-operation received from University College, London, throughout the Conference and during the weeks of organisation beforehand, are greatly appreciated by the Society.

Through the courtesy of Shell-Mex and B.P. Ltd., each delegate was presented with a handsome brief-case in blue leather bearing the crests of the Society and of the Institute in gold, in which to carry his volume of Conference papers and other material. The Council wish to thank Shell-Mex and B.P. Ltd. for their generosity and Mr. C. R. Field and Mr. E. L. Bass.

Delegates began to arrive in London the week before the Conference to attend the Society of British Aircraft Constructors Display at Farnborough. Through the courtesy of the Government and the Industry, interesting visits were arranged for the American delegates to the following firms:—

Thursday 10th September—Vickers-Armstrongs Ltd. (Weybridge) and Rolls-Royce Ltd.

Friday 11th September—Rolls-Royce Ltd.

Monday 14th September—Handley Page Ltd.

Friday 18th September—Bristol Aeroplane Co. Ltd. and Saunders-Roe Ltd.

The Wilbur Wright Memorial Lecture, usually held in May, was arranged to coincide with the Conference and was given on 14th September by Professor N. J. Hoff. This, the 41st Wilbur Wright Memorial Lecture, was followed by a Reception and Buffet Supper at 4 Hamilton Place. Attended by well over 700, including delegates, members of the Society and guests, the Reception was a welcome to the American delegates and a pleasant start to the Conference.

At the end of the Conference it was announced that the Fifth Conference would be held in June 1955 at Los Angeles.

TECHNICAL SESSIONS

THE OPENING CEREMONY

The Conference was officially opened at 9.30 a.m. on Tuesday 15th September, when delegates gathered in the large hall at Friends House, Euston Road. Sir William Farren, President of the Society, presided and was accompanied on the platform by Mr. C. J. McCarthy, President of the Institute of the Aeronautical Sciences, Professor B. J. Lloyd-Evans, Professor of Mechanical Engineering, University College, London, Mr. S. Paul Johnston, Director of the Institute of the Aeronautical Sciences, and Dr. A. M. Ballantyne, Secretary of the Society.

SIR WILLIAM FARREN: It was first his duty and pleasure to thank the authorities of University College for allowing them to use certain rooms and other resources of the College. They had intended to hold their Meetings there, but the number of members exceeded their expectations, and the College's largest Hall was too small. The Hall in which they were assembled was the Friends Meeting House. That name, he believed, would explain itself as well to their American friends as to themselves. It was a good omen for the Conference. They were fortunate in finding a hall near to University College large enough for the whole of the membership.

They had hoped that the Provost of University College, Dr. Ifor Evans, would have been there to welcome them, but unfortunately he could not come. He was represented by Professor B. J. Lloyd-Evans, Professor of Mechanical Engineering, whom he would ask to say a word on behalf of the College.

PROFESSOR B. J. LLOYD-EVANS: He found the task which fell to him a pleasant one, for he had to welcome them there. At the same time he felt a fraud because the meeting was not being held in University College. They had no room there for such a meeting because their Great Hall had been destroyed during the War and they had not yet been able to rebuild it.

He thought the atmosphere of the College a good one for such a meeting, for there was usually a feeling of friendship and intelligent enquiry as well as, he hoped, a little learning.

He had noticed that quite a number of incidents in connection with the history of the College had been recorded in the brochure, but there was one which had not been mentioned and which might amuse them. A few years ago a rather pompous professor of mathematics said he would relieve one of his assistants by taking over mathematics for engineers and decided to lecture to students in their last year. He held an examination for 30 students of that class and picked out the 15 best and gave them a lecture. The next morning he again lectured to the class, but on looking at them discovered they were not the same fifteen. After that he left his assistant to handle the class.

He sincerely hoped that the delegates to the Conference would enjoy their visit to University College.

SIR WILLIAM FARREN: He thanked Professor Lloyd-Evans. He welcomed all members of the Conference, especially their friends of the Institute of the Aeronautical Sciences.

His first duty was to read a message from Her Majesty the Queen, Patron of the Royal Aeronautical Society. He had informed Her Majesty that the Conference was to be held, and she had graciously replied.

He now called on Mr. Charles J. McCarthy, President of the Institute.

MESSAGES RECEIVED

To Her Majesty the Queen

On behalf of the Council and all the Members of the Royal Aeronautical Society, we submit to you, our Patron, our humble duty on the occasion of the Fourth Anglo-American Aeronautical Conference to be held in London from the 15th-17th September, jointly with this Society and the Institute of the Aeronautical Sciences of America.

These Conferences are attended by leading aircraft designers, scientists and research workers from both countries, to discuss aviation problems.

They were first started in 1947 and are growing in importance, and result in the discussion and better understanding of our respective problems in the field of aviation development.

Signed: WILLIAM S. FARREN.

From Her Majesty the Queen

The Queen, as Patron of the Royal Aeronautical Society, sincerely thanks the Council and Members of the Society for their kind and loyal message sent on the occasion of the Fourth Anglo-American Aeronautical Conference.

Her Majesty trusts that the joint deliberations of the Society with the Institute of the Aeronautical Sciences of America will meet with every success.

From President Eisenhower

To the Royal Aeronautical Society of Great Britain and the Institute of the Aeronautical Sciences of the United States, I am happy to send warm greetings on the occasion of their Fourth Anglo-American Aeronautical Conference.

This meeting is an example to the World of the meaning and benefits to mankind of mutual co-operation and understanding between peoples of free and friendly nations in these troubled years. Our ability to maintain our cherished freedoms and our mutual hopes for world peace increasingly depends upon such co-operative efforts.

My best wishes for a most successful Conference go to all of you.

DWIGHT D. EISENHOWER.

From R. B. Anderson, Secretary of the United States Navy

On the occasion of the Fourth International Conference in London of the Institute of the Aeronautical Sciences and the Royal Aeronautical Society I am gratified by the continued success of this joint effort.

The mutual understanding and respect fostered by the association of the aeronautical engineers of the United Kingdom and the United States contributes directly to the furthering of the science of aeronautics and the accomplishment of our mutual defence aims.

R. B. ANDERSON.

From Harold E. Talbott, Secretary of the United States Air Force

On behalf of the United States Air Force I send congratulations and best wishes for success of the Fourth Anglo-American Aeronautical Conference.

In these difficult times when air power plays such a dominant role in determining the course of world affairs, the free interchange of ideas and information between the Institute of the Aeronautical Sciences of the United States and the Royal Aeronautical Society of Great Britain is of utmost importance.

The Conference, I am sure, will do much to advance the science of Aeronautics on both sides of the Atlantic, and to further strengthen the bonds of friendship and understanding between the peoples of our two great nations.

HAROLD E. TALBOTT.

Left to right: Dr. A. M. Ballantyne, Secretary of the Society, Mr. C. J. McCarthy, President of the Institute of the Aeronautical Sciences, Sir William Farren, President of the Society, Mr. S. Paul Johnston, Director of the Institute, and Professor B. J. Lloyd-Evans, Professor of Mechanical Engineering, University College, London.

MR. CHARLES J. McCARTHY: He expressed the thanks of the American delegation for the warm welcome they had received. He also thanked Professor Lloyd-Evans and through him, University College, for the courtesy that they had extended. It seemed to him appropriate that these meetings should be held in the Friends Meeting House, because it was a gathering of friends. As a matter of fact he had felt somewhat overwhelmed with the friendship and courtesies which had been showered on them during the past week, beginning with the invitation from the Society of Aircraft Manufacturers for their Annual Dinner and for its invitation to attend the flying display at Farnborough, so giving them an opportunity of seeing the latest products of the British industry. The American delegates were also grateful to the many firms who had extended an invitation to them to visit their works and had given them very pleasant as well as instructive days.

They now came to the more serious portion of the Conference and were to devote the next few days to listening to and discussing papers which were to be presented and which he expected would be most stimulating and informative. He hoped the American delegates would be able to find a few remarks from their experience to add to what was to be said, and to discuss the papers.

He had been assured by the highest authority that when discussing papers it was not necessary to spend half the allotted time telling the speaker what a great man he was and how much they liked him. They admitted that to begin with; all the speakers were outstanding people in their field or otherwise they would not be invited to submit a paper! So he hoped they would get down to the business of discussing and let it be lively and to the point.

They were very fortunate in having messages from the President of the United States of America, the Secretary of the Navy and the Secretary of the Air Force.

SIR WILLIAM FARREN: He was sure they were all glad to have those messages. He thanked Mr. McCarthy for his generous words and not least for his remarks about the presentation and discussion of papers.

It might be as well if he admitted that the Conference clashed with one or two other aeronautical events. In England, in 1953, however, it was almost impossible to arrange anything which did not clash with something else. But they had thought that this was the best time for the Conference, because it was most likely to bring a good number of their American friends. They had been justified in that belief. There were over 150 American members. The total membership was over 700, which exceeded all previous records. There were over 300 members from the British aircraft industry, which showed the value which the industry attached to the activities of their two Societies.

The Programme was a promising one. The number of papers, twelve, was less than at any previous Conference. This reduction had been made with the full agreement of the Institute, and had reduced the time occupied by the Conference to three days. Each paper was allowed 1½ hours and there was therefore opportunity for good discussions. He would however ask each lecturer to try to limit himself to half an hour. Active discussion was the life-blood of a Conference.

He had pleasure in thanking Mr. Fields and the Shell-Mex Company for their generous gift of the folders in which they found their papers. He trusted they would find them useful after the Conference.

Mr. McCarthy had mentioned that our American friends had already seen something of British aeronautics outside the Conference. There had already been three visits under the auspices of the Conference and two more were arranged. Many had no doubt also made private arrangements. He believed that these opportunities, combined with the Conference, would make their visit to this country profitable. He could assure them it had been a great pleasure to welcome them.

To the ladies of the Conference he had already expressed a special welcome in the note which he had written at the beginning of the Secretary's booklet.

Diary of the Conference

MONDAY 14TH SEPTEMBER

Reception at 4 Hamilton Place, following the 41st Wilbur Wright Memorial Lecture by Professor N. J. Hoff.

TUESDAY 15TH SEPTEMBER

Official Opening of the Conference at Friends House, Euston Road, at 9.30 a.m.
Lectures — 11.30 - 1.00 p.m., 3.00 - 4.30 p.m.

WEDNESDAY 16TH SEPTEMBER

Lectures — 9.30 - 11.00 a.m., 11.30 - 1.00 p.m., 2.30 - 4.00 p.m.
From 4.30 p.m. - 5.30 p.m. N.A.C.A. film "Some Crash Fire Studies."
6.30 p.m. — Reception by the Government to American and Overseas Delegates at Lancaster House.

THURSDAY 17TH SEPTEMBER

Lectures — 9.30 - 11.00 a.m., 11.30 - 1.00 p.m., 2.30 - 4.00 p.m.
Closing Ceremony 4.30 p.m.
Conference Dinner and Dance at The Dorchester, London, 7.30 p.m. - 2 a.m.

In conclusion, he wished to say a word about the Secretary. He had evidently studied the methods of his predecessor, their friend Captain Pritchard, to whom Conferences seemed to come naturally—they would remember the one at Brighton. Dr. Ballantyne, aided by his devoted Staff, had been entirely responsible for the arrangements. He had produced a remarkable booklet, to which Professor Lloyd-Evans had already referred. He had been afraid that some of its references to University College might have been considered libellous, but Professor Lloyd-Evans had relieved his anxiety. Even now he was not quite sure what some of the words meant, but he had been assured that they were quite respectable, the last being a term of encouragement and joy. He felt that that was a suitable note on which to close his introductory remarks.

Tuesday 15th September

MORNING SESSION

Structures for High Speed Aircraft—H. L. HIBBARD and J. F. MCBREARTY, Vice-President Engineering, and Chief Structures Engineer, California Division, Lockheed Aircraft Corporation.

Chairman: SIR WILLIAM FARREN, President of the Royal Aeronautical Society.

This paper contained a broad discussion of some of the more important structural problems in high performance aircraft and some of the methods and materials currently available to the aircraft designer for their solution. The constant demand for refined design and advanced performance has brought new modes of failure and new prominence for classic phenomena such as thermal stresses, creep, nuclear radiation, fatigue, elastic effects and transient stresses. These phenomena were discussed and some observations on the means of coping with them were included. Man power utilisation, organisation refinements, computing machinery, flight load measurements, structural testing facilities and stress analysis data were the devices discussed from this standpoint. An imposing array of materials is at the disposal of the aircraft designer and new materials are being developed to improve structural efficiency and to help in surmounting the difficulties of higher performance. High heat-treated steel, 78ST, titanium and certain plastics were discussed briefly, and some potentialities for the future were mentioned. It was concluded that powerful tools were available to provide safe, efficient and practical structures for high performance aircraft and that structures could keep pace with developments.

AFTERNOON SESSION

Pod Mountings of Jet Engines—GEORGE S. SCHAIRER, Chief of Technical Staff, Boeing Airplane Company.

Chairman: AIR CMDRE. F. R. BANKS, Principal Director, Engine Research & Development, Ministry of Supply.

Jet engines can be mounted on large aeroplanes in a number of different arrangements including submerged wing mounting, pod mounting along the wing and body mounting. The basic problems of pod mounting jet engines were examined and the author claimed that pod mounting permitted engine performance approaching the ultimate; that the lift/drag ratio of aeroplanes with pod mounted engines are unequalled, and that methods can be found to mount the pod without deleterious effects upon critical Mach number. The structural weight of aeroplanes with pod mounted engines is good and the resulting overall range-payload performance is excellent.

Other points made were that:—(*i*) Pod mounting permits isolation of the power plants from the aeroplane and permits division of the power plants into a sufficient number of elements so that failure of one power plant does not reduce the probability of continued flight. (*ii*) The excellent engine maintenance possible with pod mounting provides increased safety, decreased maintenance costs and increased aeroplane utilisation. (*iii*) Pod mounting provides a number of by-products in the form of improved stability and control, decreased noise, increased flexibility for future growth and reduced cost.

Wednesday 16th September

MORNING SESSIONS

Power Plants for Rotary Wing Aircraft—A. GRAHAM FORSYTH, Chief Helicopter Engineer, Fairey Aviation Co. Ltd.

Chairman: W. LITTLEWOOD, Vice-President Engineering, American Airlines Inc.

The paper opened with a suggested revision of engine and turbine test conditions to cover helicopter requirements. This was followed by a description of a flat engine which the author considered would improve most installations. A number of diagrams showed the engine installed in various types of aircraft.

The change-over from piston engines to turbines was discussed and covered by diagrams.

The major portion of the paper covered experimental work on the development of jet-driven rotors. The application of pulse, ram and pressure jets was fully covered and, in addition, these systems as applied to large aircraft were described and illustrated. A description of the Fairey "Gyrodyne" was included.

Various methods of providing the air supply for pressure jets were shown and a pneumatic system which can be applied to any form of rotor or propeller drive was also included. French helicopters with jet-driven rotors were also considered.

Structural Adhesives for Metal Aircraft—N. A. DE BRUYNE, Managing Director, Aero Research Ltd.

Chairman: PROFESSOR N. J. HOFF, Head of Department of Aeronautical Engineering & Applied Mechanics, Polytechnic Institute of Brooklyn.

This paper began with a comparison of the merits of adhesive bonding of light alloys with riveting. It described typical applications and methods of surface preparation, applications of pressure and heat and inspection procedure. A brief account was given of the strength of glued joints as influenced by their geometry and of the fundamentals of adhesion. There was also a section on the "Redux" process. Special consideration was given to sandwich structures with cores of aluminium honeycomb and the considerable advance in structural efficiency that can be

obtained was demonstrated by test results in comparison with stringer stiffened skins designed for optimum efficiency.

The Control of Flight—P. R. BASSETT, President Sperry Gyroscope Corporation.

Chairman: G. R. EDWARDS, Managing Director (Aircraft Division), Vickers-Armstrongs Ltd.

This was a review of the development of the control of flight through the past fifty years. Within the scope of this subject was included the human pilot, all the instruments which aid him, and the automatic pilot. The human pilot as the preceptor and director part of the servo-control system of an aircraft was quite sufficient until he started to fly across country at altitudes of several hundred feet. Then he needed supplementary instruments. Tachometer, altimeter, air speed indicator and magnetic compass were the first group added. After the First World War the essential need to accomplish blind flying caused the addition of turn-and-bank indicator and rate-of-climb indicator. In 1929 the gyro horizon and directional gyro simplified blind flying. During the decade of the 1930's, with adequate flight instruments, automatic pilots, and radio aids to navigation, the world-wide systems of air transport grew rapidly. Since the Second World War emphasis has been placed on instrument approach development. The result has been a new type of instrument called a flight indicator, which indicates to the pilot how he is to move his controls. Instrument landing was considered on the basis of achieving the maximum percentage of successful approaches rather than attempting to accomplish actual contact by instrument. It was conceded that the final contact could be made much easier by normal vision, provided that the pilot was assured of time for transition from instruments to vision of the ground and of being at the proper position at that time.

The first standardised flight panel has been approved by United States air transport companies and the Civil Aeronautics Board.

The Aerodynamics of Compressor Blade Vibration—H. PEARSON, Chief Research Engineer, Rolls-Royce Ltd., Derby.

Chairman: DR. H. L. DRYDEN, Director, National Advisory Committee for Aeronautics.

The paper considered the causes of compressor blade vibration and showed that, in the unstalled region, damping was almost entirely aerodynamic, internal damping of the blade material and damping at the blade root being negligible. When wakes were present, the vibration of the blade, which was generally of the fundamental flexural type, might be assessed from the aerodynamic excitation and damping. Such considerations led to simple expressions for blade stresses.

A prime cause of blade failure was flutter when the stage was stalled. The high stresses set up in this condition were due to the absence of any degree of damping and a modest degree of "built-in-" damping would reduce them to safe limits. A comparison was made between tests on cascade rigs and rotating stages, and it was concluded that much more work was required on this subject. The phenomenon of the rotating stall, which complicates this comparison, was also described.

A brief consideration was given to torsional vibrations. So far, few failures had been ascribed to such vibrations.

AFTERNOON SESSIONS

Aircraft De-Icing by Thermal Methods—J. L. ORR, D. FRASER and J. H. MILSUM, Low Temperature Laboratory, National Aeronautical Establishment of Canada.

Chairman: DR. T. P. WRIGHT, President, Aeronautical Laboratory, Cornell.

With increasing flying speeds, the icing problem becomes more critical and, where thermal anti-icing methods of protection are used, the quantities of heat required become prohibitive. If, instead, a thermal de-icing principle is employed whereby the ice is shed in the solid state by periodically heating the surface on which it forms, a reduction in thermal energy requirements can be achieved.

The various considerations involved in the selection of a protective system and the characteristics of ice formations as they affect de-icing were discussed. The general principles of a thermal de-icing system were outlined and the application of electro-thermal de-icing to propellers, wings and helicopter rotors was described. The performance characteristics of thermal de-icing systems were considered and a practical example of their application to a turbo-prop transport aircraft was worked out to provide power and weight data. It was concluded that electro-thermal de-icing methods of protection might be applied with advantage to many of the icing problems of aircraft.

Some Factors Affecting Helicopter Design and Future Operations—R. H. MILLER, Vice-President Engineering, Kaman Aircraft Corporation.

Chairman: E. MENSFORTH, Chairman, Westland Aircraft Ltd., and Normalair Ltd.

Some of the problems encountered with various rotor systems designed to increase helicopter utility, primarily as a means of commercial transport, were discussed. Promising means of increasing the cruising speed, such as the partially unloaded rotor, the rotor with offset flapping hinges, and the supersonic rotor, were evaluated briefly, together with the weight penalties associated with each configuration. Various means of rotor propulsion, and their relative limitations and advantages, were evaluated as a function of range and weight reduction. The problems involved in establishing adequate structural design criteria were briefly discussed, together with the results of recent tests aimed at a better understanding of the harmonic blade loading problem. Finally, the handling qualities of conventional helicopters were analysed and means were suggested for providing satisfactory and reliable blind flying characteristics for all-weather conditions.

Some Crash Fire Studies—N.A.C.A. Film presented by DR. H. L. DRYDEN.

Thursday 17th September

MORNING SESSIONS

Some Recent Advances in Boundary Layer and Circulation Control—COURTLAND D. PERKINS and DAVID C. HAZEN, Professor and Assistant Professor, Dept. of Aeronautical Engineering, Princeton University.

Chairman: A. A. HALL, Director of the Royal Aircraft Establishment.

A brief description of boundary layer control systems utilising blowing slots, suction slots or porous material was given. The differences between such systems which are

primarily designed to give control over separation and hence low drag, and those systems using blowing or suction over flaps or employing trailing edge suction slots to change the circulation, and hence the lift at a given angle of attack, were pointed out.

A discussion of an experimental investigation conducted by the State College of Mississippi utilising a sailplane to examine the effects of distributed suction on the stabilisation of the laminar boundary layer was presented. It was shown that by tailoring the porosity of the surface, so that just sufficient suction was applied to maintain the laminar boundary layer without excessive thinning, the lowest total drag (aerodynamic plus suction power required) could be obtained.

To demonstrate the difference between boundary layer control and circulation control, the results of an experimental and theoretical programme conducted by Princeton University investigating the possibilities of trailing edge suction slots were reported. The importance of the location of the trailing edge stagnation point was discussed and a theoretical technique for predicting the aerodynamic characteristics of a profile equipped with such a slot was presented and compared with experimental results.

The use of a "snow cornice" type shape, in which it is possible to entrap a vortex either automatically or by means of suction, was reported. Certain applications to wind tunnels and wings which had given promising results were discussed. It was felt that this use of the trapped vortex represented a major advance and that considerable increases of efficiency in all subsonic expansion problems might be possible by utilising this principle.

Basic Pre-requisites For Production—R. A. NEALE, Vice-President Manufacturing, Canadair Ltd.

Chairman: C. E. ROBERTS, Washington Representative, Boeing Airplane Company.

Basic pre-requisites for production were examined with particular reference to the experience of Canadair Ltd. with the production of F-86 Sabre fighters and the T-33 and T-36 training aircraft. The subject was considered under the four headings of Plans, Plant, Policy and People.

Plans must cover every phase of production, telling who is going to do what and when and how often. Gaps in information must be filled with the best estimates that can be made. Examples of the various charts and schedules necessary for production planning were discussed in detail.

The fitting of the plant and policy to the plan, and the interaction between these three factors and the human factor was shown to be of prime importance, if economic production on schedule were to be achieved.

Some Observations on the Problem of Fatigue of Aeroplane Structures—R. V. RHODE, Assistant Director for Research, National Advisory Committee for Aeronautics.

Chairman: B. S. SHENSTONE, Chief Engineer, British European Airways.

The problem of fatigue of aeroplane structures involves two essential elements: first, the start of a fatigue crack during the useful life of the machine, and second, the undetected growth of the crack until the structure can no longer support the normal operating loads.

Consideration of design trends over the past twenty years, coupled with a simple analysis of the relations between gust intensity, load factor and stress, indicates that the number of miles of flight before a fatigue crack may appear in the primary structure of aeroplane wings has decreased markedly over the years. This tendency was shown to be associated primarily with improvements in static structural efficiency; other trends in design or performance, such as in the wing loading, load factor, cruising speed, aeroplane size, dynamic response, and so on, were shown or stated to have been favourable, of secondary importance, or approximately compensated by

Some of the American delegates photographed during the visit to Vickers-Armstrongs Ltd. on 10th September.

related effects. It was indicated that jet transports probably would not enjoy much relief from fatigue as a result of reduced gust frequency at the higher altitudes.

Attempts to solve the problem by estimation of the fatigue life, by the provision of fatigue damage, or crack indicators, or by attention to detail design, were discussed.

The extent of fatigue cracking and loss in strength was discussed in relation to crack detectability and structural redundancy. It was argued that redundancy does not necessarily provide safety, but that structural arrangements having intrinsic safety, by virtue of detectable appearance of cracks before serious loss in strength occurs, were not only possible, but already existed. The opinion was expressed that provision of intrinsically safe structures was of as great importance as estimation of the fatigue life.

AFTERNOON SESSION

The Introduction of the Comet into Service—A. C. CAMPBELL ORDE, Operations Development Director, British Overseas Airways Corporation.

Chairman: C. J. MCCARTHY, President, Institute of the Aeronautical Sciences.

British Overseas Airways' experiences with the de Havilland Comet air liner during the planning, trial and early operational stages, was described and conclusions were drawn for future operations with jet aircraft.

To provide a comparison with a conventional aircraft the Comet was compared with the Argonaut (Canadian-built DC-4) and it was shown that all the expected advantages of higher block speed were realised. The causes for all irregularities of service during the first year were analysed and discussed.

The operating procedures used by slower aircraft were shown to put a heavy load upon the Comet crew and it was suggested that new methods and equipment should be developed to lighten the work of the crew and ground staff.

It was concluded that by the results of the first year of operation the inherent advantages of the type of aircraft had been demonstrated and that in future operators must define their requirements precisely and must establish close partnership with the designers and manufacturers if the utmost efficiency were to be obtained.

CLOSING SESSION

THE CLOSING CEREMONY

The closing session of the Conference was presided over by Sir William Farren, President of the Society. He was accompanied on the platform by Mr. C. J. McCarthy, President of the Institute of the Aeronautical Sciences, Brigadier E. Gueterbock, Secretary of University College, Mr. S. Paul Johnston, Director of the Institute and Dr. A. M. Ballantyne, Secretary of the Society.

SIR WILLIAM FARREN: Apart from the informal occasion arranged for that evening, they had finished what they had set out to do. They had been most hospitably welcomed at University College, and in the absence of Professor Lloyd-Evans, whom they had met at the opening of the Conference, Brigadier Gueterbock, the Secretary of the College, had come to address them.

BRIGADIER E. GUETERBOCK: Theirs was the largest of the Colleges of the University of London and they had no special connection with local industries, as happened in the case of so many provincial universities. They sent their students all over Britain—indeed all over the world—and within their walls they tried to give the students a broad education. Students of the arts, the sciences, laws, medicine and engineering all mingled together, and the four engineering departments played a very important part in the life of the College.

It was their belief that there should be a close connection between universities and industry, and they hoped that they helped industry by sending out graduate recruits and also by research. In the matter of research they welcomed the advice of industry and liked to find out about the problems that beset it and the lines along which industry was interested to see development.

At University College they were particularly strong on the post-graduate or research side. Out of a total of 3,500 students more than 750 were engaged on research, which was a very high proportion, and if any firm would like to send some of its research staff for a year or so to work with them they would be only too glad. He hoped and believed that such people would get encouragement from meeting the academic staff, and the latter certainly would receive inspiration from men with practical experience.

The College was particularly glad to welcome the leaders of the great aircraft industries of the United States and Britain, and hoped that they might have the pleasure of doing so again, if not collectively then individually. Possibly some members of the Conference would find time to visit the College during term when the laboratories would be full of students; with that hope his last words were not "Good-bye" but "Au revoir."

SIR WILLIAM FARREN: He thanked the College, through the Secretary, for all its generous help. He felt sure that the general view of members was that this Conference had been successful. He regretted the trouble with the loud-speaker equipment. An unforeseeable failure of the electronic equipment to come to the aid of those speakers who could not cope with a room of the size of Friends House almost frustrated one of the main purposes of the Conference.

He was reminded of a story which Dr. von Kármán had told him on that topic. "Do not forget that these electronic machines, including the calculating machines which so many of you quite rightly admire, and the guided missiles you are dreaming up, can work just as well with the wrong information. They can go mad, and you will not be able to tell that they have gone mad until afterwards."

They had got off to a good start with Mr. Hibbard's paper which was a forthright statement of achievement, and they had certainly finished well with Mr. Campbell Orde's masterly paper and the ensuing excellent discussion.

It appeared that there was no easy solution to the problem of the allocation of time between the reading and the discussion of papers. In the end, it had been left to the authors to settle for themselves. Short of

having a Conference of indefinite length, it was difficult to see what more could be done. It appeared, however, that the policy of having relatively few papers, and allowing a full hour and a half for each, was the best of the available alternatives.

He had been cheered to see many energetic private meetings going on in all corners of the buildings. He felt that one of the main purposes of the Conference had undoubtedly been achieved. There had been everywhere a pleasant atmosphere of cheerfulness and warm friendliness. Speaking for the Council, for the Secretary and for the Staff, they had done their best and, having learned a great deal, hoped to do better next time.

MR. CHARLES J. McCARTHY: His only regret was that the Conference had come to an end, although he did not know how many more days of such intense activity, both night and day, they could have survived. It had been a very successful meeting and they owed a great deal to the hospitality which had been extended by the members of the Royal Aeronautical Society.

Speaking on behalf of all the American delegates he thanked the speakers and the people who had discussed the papers for the effort which they had put into it. The Conference had been very rewarding and they had all learned a great deal. He wished also to extend their thanks to Dr. Ballantyne and his very hard working staff, who had written thousands of letters and had ensured that every delegate had one in his hands at the right time so that he knew exactly where to go. The staff had done an outstanding job in controlling such a complicated operation.

In general, in his opinion the Fourth Conference had been the best so far. The child had now been brought to a fairly good size and was a growing and lusty infant, and so it was essential that the series of conferences should be continued and should improve as time went on. Speaking on behalf of the Institute of the Aeronautical Sciences it was his happy duty and pleasure to extend a cordial invitation to the Royal Aeronautical Society to visit them in America in 1955. There had been a meeting between the Councils of the two Societies and it seemed that the latter part of June 1955 was the period which was most convenient to the most people, and so they invited the members of the Royal Aeronautical Society to visit Los Angeles, California, on or about the week beginning 21st June 1955, where they would do their utmost to return the hospitality so generously extended to them in this country. They would do their best to give their visitors an interesting time and would arrange trips to nearby aircraft activities; they had been assured by Dr. Dryden that he would extend an invitation to the delegates to visit the Ames Aeronautical Laboratory, near San Francisco.

SIR WILLIAM FARREN: He would not hesitate, as the matter had been discussed by the Council formally and informally, on behalf of the Society to thank Mr. McCarthy and the Institute of the Aeronautical Sciences and to accept with great pleasure their generous invitation. The Society would look forward to June 1955.

THE CONFERENCE DINNER-DANCE

The final event of the Conference was a Dinner-Dance held at the Dorchester on Thursday, 17th September, and attended by more than 500 delegates and their ladies. Delegates and guests were received by Sir William Farren, President of the Society, and Miss Mary Farren and by Mr. Charles J. McCarthy, President of the Institute, and Mrs. McCarthy.

The following is a report of the speeches made at the Dinner.

SIR WILLIAM FARREN: The toasts of Her Majesty The Queen and The President of the United States of America having been honoured, they would see from their programmes that he had a very pleasant task to perform—to propose the toast of the Institute of the Aeronautical Sciences. Before that, however, he had one or two other pleasant things to do.

They had drunk to the toast of The Queen, the Patron of the Society. It was, he felt, appropriate now to tell them that His Royal Highness the Duke of Edinburgh had accepted the Honorary Fellowship of the Society. They would welcome him not only because of the lively interest he displayed in all aeronautical matters, but all the more warmly because of the pleasure he found in flying.

Many of their friends from America were present at the Wilbur Wright Lecture, and they would remember that he then had the pleasure of presenting the Society's Honours to certain members. One of the three Honorary Fellows had been unable to be present then, but they were happy to have with them that evening Lord Hives of Duffield and he invited him to accept the scroll of the Honorary Fellowship of the Society.

He would take the opportunity of reading to them what was written on the scroll:

"The Society—established for the General Advancement of Aeronautical Science and Engineering and more

Left to right: Sir William Farren, Miss Mary Farren, Mrs. C. J. McCarthy, Mr. C. J. McCarthy.

particularly for promoting that Species of Knowledge which distinguishes the Profession of Aeronautics."

He did not know whose words they were, but he could not think of any which would express more concisely or more completely the objects for which their two Societies existed.

The Dinner-Dance was a domestic affair, almost a family affair. He had however invited a few guests, chosen because frankly he thought they would like to have them. Major General Francis Griswold, whom they regarded almost as a fellow countryman, was specially welcome. Air Marshal Sir John Boothman had long been a good friend to them and to British Aviation, and he was sure all members of the two Societies would join with him in wishing him good fortune in his new appointment to Coastal Command. Lord Douglas of Kirtleside he welcomed both for himself and as Chairman of British European Airways. He was particularly glad that Mr. Hugh Burroughes, President of the Society of British Aircraft Constructors, to whose co-operation and support they owed so much, had been able to come, and he welcomed for personal reasons Sir Frank Spriggs, Managing Director of the Hawker Siddeley Group.

He could not close this part of what he had to say without welcoming the ladies. Without them the dinner would have been dull and the dance unthinkable. In his introduction to the Conference Booklet he had expressed the hope that the ladies from America would enjoy themselves in London. He felt sure they would look back on this evening with pleasure.

It was next his privilege to propose the toast of " The Institute of the Aeronautical Sciences." Their American friends had come here at a time when the country stopped work for a moment in order to have a look at flying. Many had come for what was now almost an institution—Farnborough Week. To most of England Farnborough now meant flying. But it had meant flying to many present that evening long before the days of the S.B.A.C. Display. It had meant flying to him as long ago as 1915. They had seen something of our latest efforts, and he hoped they felt that we were doing our share in advancing the art.

In a leader in that day's *Times* he had read that the world in which we live is now " The Airman's Globe " and that upon the imaginative planning of the great air routes and ports will depend the fruitful development of all its life. The responsibility on us was great. That afternoon's paper on the first year of the Comet's operations, and the generous tributes paid in the discussion to all who had borne the burden, was a good augury for a real understanding between us, in England, in the Commonwealth, and in America. It had seemed to him like a look into the History of the Future.

He was interested in history, particularly that of America. Our young people knew little of it and should learn more. They knew about 1066 and " all that," but were likely to confuse the American Civil War with the War of Independence! In January when he was in Washington, an Englishwoman had said that her children there seemed rather ahead of American children of the same age, and were not getting much out of school. Would they lose by a year or two in America? How could they best use their time? He had said: " Let them learn something of American history from Americans, and so come to understand something of America, and the ways and outlooks of Americans—as I have tried to do for 25 years."

He had read that the Prime Minister—and he would remind them that Sir Winston Churchill was an Honorary Fellow of the Society—had said to Mr. Adlai Stevenson that " America saved the World." He should know. We had done our part. The flight over London earlier that week of aircraft of Fighter Command of the Royal Air Force was a symbol—to remind us of the Battle of Britain, and of what those who fought it did to help to save the world when it was in its greatest danger. That was only the beginning of our effort, though it was possibly flying's most significant contribution.

They recognised the achievements of America in all ways in peace and in war, and that evening they paid a tribute to her achievements in aeronautics. He saw no future for the civilised world except in friendship and alliance with them, of which the relations between the Society and the Institute were a vital part.

This relationship had become closer and closer over the years, and the Joint Conferences had strengthened it. When he drank to the toast to which he was about to ask the members of the Society to rise, he would think rather of the future, and of the young men who would make up their membership. It was the friendship between them and their fellows over here which would build a safeguard for the future of civilisation. It would be achieved only by their joint efforts, and upon its integrity all else would depend.

He did not think of that safeguard as a high wall behind which they and theirs might live in blind trust. He thought of it rather as a living impalpable curtain, composed of their spirit, their watchfulness, their imagination and their courage.

The toast was—" The Institute of the Aeronautical Sciences," which he coupled with the name of his friend Mr. Charles J. McCarthy, the President of the Institute.

MR. CHARLES J. McCARTHY: It was his privilege and pleasure to be there on that happy occasion and to be the representative of the delegates of the Institute of the Aeronautical Sciences and have an opportunity of expressing, on their behalf, their great appreciation of the kind sentiments which had been expressed by Sir William Farren. They had had a thoroughly enjoyable visit and were greatly indebted to their hosts for the hospitality which had been showered upon them in so many ways. They had certainly enjoyed the meetings and the visits. He was sure the present had been the best of the whole series of Conferences. The only complaint, if any, might be that while they had had the opportunity of meeting many friends, renewing old acquaintanceships and starting new ones, the time had been too short to do all the things they would have liked to do or to accept all the invitations extended to them.

In general he thought the particular object of the Conference had been accomplished and he thought that was well expressed in the message which they had received from the President of the United States. With their indulgence he proposed to read part of it to them:—

> "This meeting is an example to the world of the meaning and the benefits to mankind of mutual co-operation and understanding between peoples of free and friendly nations in these troubled years. Our ability to maintain our cherished freedoms and our mutual hopes for world peace increasingly depends upon such co-operative efforts."

He was sure their efforts had been most co-operative and thanks to the casual and friendly atmosphere created by the Chairman the meetings had been very fruitful. They had all benefited from listening to the talks and discussions and the visits to Farnborough and the other activities in which they had been able to take a part. In addition to all they had gained from the Conference, he could not forget the fact that in America they had profited over the years from British effort and inventiveness. Thinking back a few years the names of Lanchester, Melvill Jones and Bairstow came quickly to mind and one recalled their important contributions to the aeronautical sciences. He believed Richard Fairey invented a landing flap for aircraft, but he was so well ahead of his time that by the time the Industry was ready to make use of the device his patents had long expired. There were so many other things which they had been able to use, but he could not fail to mention Sir Frank Whittle who had pushed them into the jet age and had created so many problems for the engineer and the aircraft designer that he did not know whether they should thank him or not. He was sure that the numerous engineering ideas that had been transported back and forth across the Atlantic had been to the benefit of them all.

When he had arrived in England he had been met by Sir William and the first thing he had asked was "What have I got to do?" "You have nothing to do," Sir William replied, "I have nothing to do. Ballantyne has done it all." At the time he was not sure whether he was being given facts or not, but time had proved that he was completely right. Dr. Ballantyne like the good executive he was, had not done it all himself. He had been well and ably served by an efficient staff and he would like particularly to pay tribute to them. They had made a very complicated and difficult task look quite easy and they certainly had smoothed the visitors' way and the delegates had profited greatly by their kind assistance.

It was now his pleasant duty to talk about what was going to happen in the future.

They were now closing the Fourth Conference and were looking forward to the next. They could not look forward more than one meeting at a time, but they hoped there would be a long series to come. He was happy to say that the two Councils had agreed that the venue for the next Conference would be in Los Angeles, California, in 1955. They hoped that it would be convenient to a large number and invited all the members of the Royal Aeronautical Society to come and urged that as many as possible should persuade their wives to join them. They would do their utmost to maintain the high standard which had been established n the past week or so and looked forward to their presence two years hence.

He suggested that the Americans present should drink to the health of the Royal Aeronautical Society with particular recognition of the great friendship and hospitality which they had received from the members.

SIR WILLIAM FARREN: He thanked Mr. McCarthy for the Institute's invitation to the next Joint Conference, in Los Angeles in 1955. The Society would give its wholehearted support, and do everything possible to ensure that the number of members from the Country was worthy of the occasion. It would provide a valuable opportunity for visiting American aeronautical activities on the West Coast, of which he hoped the British Aircraft Industry would take full advantage.

Following the Dinner there was an unusual and enjoyable cabaret of English folk dancing and Scottish reels performed by the Ravensbourne Morris Team and the Royal Scottish Country Dance Society. The Scottish dancers were accompanied by a piper and were compèred by the Secretary. Dancing was then resumed and continued until 2 a.m. So ended the Fourth Conference.

The Society wishes to place on record its appreciation of the courtesies extended to delegates by the Government, the Aircraft Industry, the Society of British Aircraft Constructors, University College, London, Shell-Mex and B.P. Ltd., and to thank all those who, by their willing co-operation and assistance contributed to, and made possible, the success of the Fourth Anglo-American Aeronautical Conference.

TECHNICAL NOTES

This section of THE JOURNAL *is intended for short papers, comments upon published or unpublished papers with a limited circulation, interim results of research in hand, or notes for further research, or comments on notes already published, or letters raising technical points or asking questions. Contributions, which will be eligible for Journal Premium Awards, will normally be published within two months of being received.*

The Noise from Aircraft Travelling with Velocities Greater than that of Sound

by

A. G. WALTERS

(Armament Research Establishment, Fort Halstead)

THERE has been much discussion of late on the nature of the noise from aircraft travelling with velocities greater than that of sound. It is possible that these phenomena can largely be explained in terms of the Doppler effect. For a source of sound travelling with a velocity greater than that of sound relative to the observer, the Doppler effect takes a very interesting form. It is discussed mathematically in Ref. 1.

To illustrate the effect Fig. 1 is constructed as follows:—

The source of sound starts from the point O and moves in a direction OS with constant velocity V so that at a time t later the source is at S where $OS=Vt$. The sound pulse which is emitted from the source at $t=0$ moves outwards with the velocity c and at time t lies on the sphere of centre O and radius ct. Sa and Sb are two generating lines of the tangent cone from S on to the sphere.

If the source of sound emits a pure note (*i.e.* of one frequency), the observed sound effects are as follows:—

(*i*) If the observer is at a point P_1 ahead of the Mach cone, no sound is heard.

(*ii*) As the Mach cone passes over the observer at point P_2, two very high frequency notes are heard simultaneously which rapidly reduce in frequency as the source moves onwards.

(*iii*) If the observer is at a point P_3 inside the Mach cone, two frequencies are heard simultaneously which reduce to two limiting constant values.

(*iv*) At a point P_4 inside the expanding sphere, only one frequency is heard, this being the normal low receding Doppler frequency.

An observer at rest hears the effects (*i*), (*ii*), (*iii*) and (*iv*) in that order. The two frequencies observed in stage (*ii*) as calculated on linear theory are both infinite.

If the velocity of the source is just less than that of sound, the intensity of the sound falls off rapidly at short distances ahead of the source, due to the fact that the path length of the sound waves reaching the observer is large. Thus the observer first hears the sound when the source is very near and in this stage the frequency is the normal high velocity Doppler approach frequency. This is rapidly followed by the low receding Doppler frequency. Thus the main effect heard by the observer is a high frequency pulse of short duration. This can be detached from the source, for example if the source is suddenly stopped the pulse continues to move forward with the velocity of sound.

If the source initially travelling with a velocity less than that of sound accelerates to a velocity greater than that of sound, the effects heard by the observer depend on his position and the acceleration. To simplify the illustration, suppose that up to the point O in Fig. 1, the source has a subsonic velocity and at the point O it suddenly attains the supersonic velocity V. At O the wave front of sound can be considered as plane. If the observer is behind the conical surface formed by rotating the generating lines Oa_1 and Ob_1 about the axis OS, he hears the subsonic Doppler effect and the

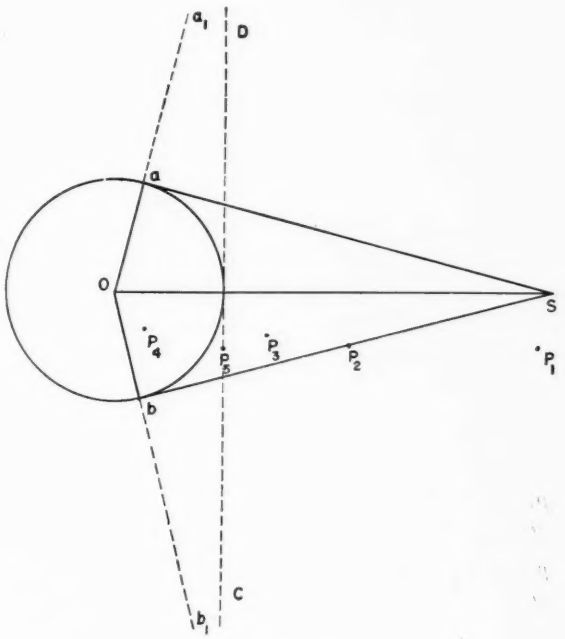


FIGURE 1.

Received July 1953.

one supersonic Doppler frequency when the sphere expands over him as in stage (*iv*). If the observer is ahead of this conical surface, he hears in turn the effects (*i*), (*ii*), (*iii*) and (*iv*) described earlier, but in addition as the plane surface *CD* passes over him (*i.e.* the observer is at the position P_5) he hears the high frequency subsonic pulse described earlier. It is easily seen that the angle contained between the lines *OS* and Oa_1 is $\cos^{-1}(c/V)$. Due to the finite value of acceleration the generating lines Oa_1 and Ob_1 form a cusp, the vertex being at *O*, the point where the source velocity becomes sonic.

The mathematical results as given in Ref. 1 are reproduced here. Let X and ρ be the co-ordinates of the observer (assumed at rest) relative to the source *S* at a given time t. That is, X is the distance measured along *SO* and ρ the distance perpendicular to *OS* (Fig. 2). Suppose the source of sound emits a sound wave

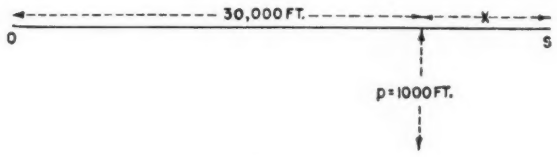


FIGURE 2.

given by $A\cos pt$. If the observer is inside the expanding sphere, he hears the one sound wave given by

$$y=\frac{A\cos p\delta_1}{\alpha},$$

where

$$\delta_1=t-\frac{XV}{(V^2-c^2)}+\frac{L_2}{(V^2-c^2)^{\frac{1}{2}}},$$

$$L_2=\left(\frac{c^2X^2}{V^2-c^2}-\rho^2\right)^{\frac{1}{2}},$$

and

$$\alpha=4\pi c\,(V^2-c^2)^{\frac{1}{2}}L_2.$$

If the observer is inside the Mach cone but outside the sphere, he hears the two sound waves given by

$$y=\frac{A\cos p\delta_1}{\alpha}+\frac{A\cos p\delta_2}{\alpha},$$

where

$$\delta_2=t-\frac{XV}{V^2-c^2}-\frac{L_2}{(V^2-c^2)^{\frac{1}{2}}}.$$

The frequency of the note emitted by the source is $p/(2\pi)$ cycles per second. The frequency heard by the observer inside the sphere is $p_1/(2\pi)$, and inside the cone he hears the two frequencies $p_1/(2\pi)$ and $p_2/(2\pi)$, where

$$p_1=p\frac{d\delta_1}{dt},\quad p_2=-p\frac{d\delta_2}{dt}.$$

Thus

$$\frac{p_1}{p}=1-\frac{V^2}{V^2-c^2}+\frac{1}{(V^2-c^2)^{\frac{1}{2}}}\frac{c^2XV}{L_2(V^2-c^2)},\quad L_2>0,$$

$$\frac{p_2}{p}=-1+\frac{V^2}{V^2-c^2}+\frac{1}{(V^2-c^2)^{\frac{1}{2}}}\frac{c^2XV}{L_2(V^2-c^2)},\quad L_2>0,$$

since

$$\frac{dX}{dt}=V.$$

TABLE I

Time (*sec.*)		*Sound heard* (*cycles/sec.*)	
$t<20{\cdot}618$		No sound	
$=20{\cdot}618$		Two very high notes	
20·743		847	2,011
20·868		548	1,709
21·118		370	1,534
21·368		308	1,471
21·618		277	1,440
22·618		235	1,398
25·618		216	1,380
27·2727	Sonic pulse	213	1,376
27·288		212	High frequency note cuts out
$t>27{\cdot}288$		212	

As an example, suppose that the aircraft travelling with subsonic velocity suddenly attains the velocity 1,500 ft./sec. at the point *O*. Suppose that the observer is at the distance 30,000 ft. from *O* as measured along *OS* and 1,000 ft. perpendicular to *OS* (Fig. 2).

Taking the origin of time as that instant at which the aircraft is at *O* and taking the velocity of sound as 1,100 ft./sec., the observer hears no sound until the time 20·618 seconds. At this time he hears the two very high frequency notes. Suppose that the frequency of the note emitted by the source is 500 cycles/sec., the frequencies heard by the observer at subsequent instances are given in Table I.

It is noted that in this example the high frequency note is maintained for 6·67 seconds. If the aircraft had attained the speed of 1,500 ft./sec. at an infinite distance away, the high frequency note would be maintained until the aircraft is beyond earshot, the final frequency being 1,375. If on the other hand the observer is near to *O*, the high frequency note persists only for a short time. For example if he is 5,000 ft. from *O* as measured along *OS*, the high frequency note persists for 0·68 seconds and terminates at the frequency 1,500.

The sonic pulse discussed earlier reaches the observer after 27·273 seconds and appears as a light

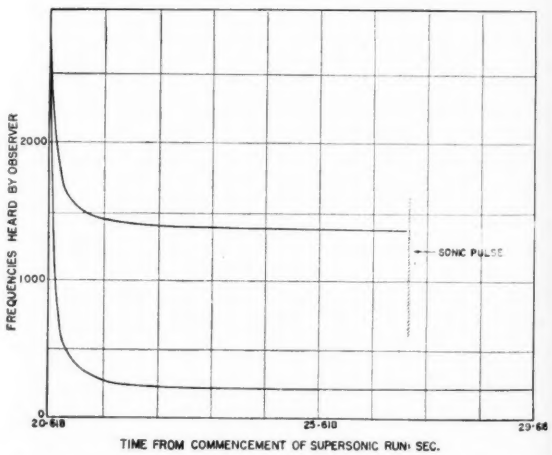


FIGURE 3. Frequency of sound from aircraft is 500 cycles/sec. Aircraft speed is 1,500 ft./sec. Observer is 30,000 ft. from beginning of supersonic run and 1,000 ft. from the line of flight.

band due to dissipation with distance. In this example the frequency heard by the pilot is taken to be 500 cycles/sec. If it is n, then these frequencies must be multiplied by $n/500$. The results are presented graphically in Fig. 3.

The aircraft as a source of sound is a complex process. The sound arises from the jet engines, from the aerodynamic causes and possibly the vibration of the airframe components, the whole forming a wide spectrum of sound frequencies, possibly containing discontinuities. Each individual frequency is "Dopplerised" in the manner shown, the whole being superimposed. Thus the effect heard by the observer comprises two separate spectra, both initially in the high frequency range and reducing rapidly with time, thereafter one continuing as a constant band frequency and the other cutting off after a short time.

The time of arrival of the sonic pulse depends on the acceleration of the aircraft. The finite acceleration

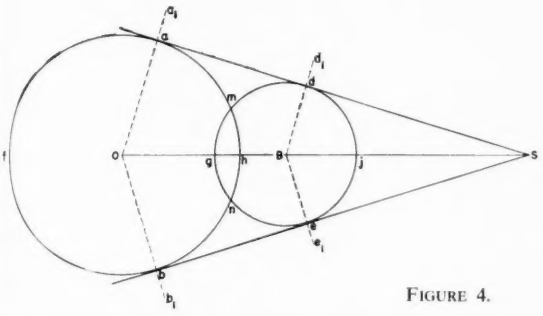


FIGURE 4.

modifies the picture, but the basic qualitative features should remain unchanged. It is known that the noise from the jet is directional, particularly in the high frequencies, but this merely changes the polar intensity diagram.

In the equations given earlier it is noted that the amplitude of the waves are each inversely proportional to L_2. This term L_2 is zero on the generating lines of the cone and thus the intensity of the waves is theoretically infinite on the surface of the Mach cone, decreasing rapidly on the inside of the cone. It is difficult to predict the sound effects heard from a high frequency spectrum which is rapidly decreasing in frequency and intensity with time. The lower frequency spectrum reduces to audible frequencies before the higher frequency spectrum. This separation increases with distance from the line of flight; in the foregoing example it is small, being of the order of half a second. If the separation is sufficient two distinct bangs are heard and these are followed by the sonic pulse. Otherwise the two bangs merge into one. The viscosity of the air has a marked effect on these high frequencies and "smooth out" the bangs in the same manner as the "crack" of a gun or of an exploding shell is heard as a "boom" at large distances.

One variant of this problem is of particular interest. Suppose that the aircraft starts its supersonic run at O and moves with constant velocity V for a time T sec. after which time it is at B (Fig. 4). At this time suppose that it suddenly decelerates to a velocity below that of sound. To illustrate the effects after t seconds from the start of the supersonic run, t being greater than T, we construct Fig. 4. With centre O construct the sphere of radius ct. With centre B construct the sphere of radius $c(t-T)$. S is the position the aircraft would have occupied if it had maintained the velocity V. Actually the aircraft is at some point A on OS inside the sphere of centre B at time t. It is easily seen that the generating lines of the hypothetical Mach cone are tangential to both spheres. The following results can be proved mathematically, δ_1 and δ_2 having the same meaning as before, X being measured backwards from S as in Fig. 2.

(*i*) In the region of space in which the inequality $0 < \delta_1 < T$ is satisfied, the lower frequency spectrum generated in the supersonic run is heard.

(*ii*) In that region of space in which the inequality $0 < \delta_2 < T$ is satisfied, the higher frequency spectrum generated in the supersonic run is heard.

The first inequality is satisfied in that region of space inside the Mach cone which is behind the spherical surface *dge* and in front of the spherical surface *afb*, that is, that region inside the surface *egdafbe*. The second inequality is satisfied inside that region of the Mach cone which is behind the spherical surface *ejd* and in front of the spherical surface *ahb*, that is, the region inside the surface *ejdahbe*. Thus in that region inside the sphere centre O which is not included in the second sphere, only the lower frequency spectrum is heard, and in that region inside the sphere of centre B not included in the other sphere, only the higher frequency spectrum is heard. In that region common to both spheres, *ngmhn*, neither spectrum is heard. In the region *amda* and *bneb*, both spectra are heard simultaneously and in the region of the cone ahead of the surface *dje* no sound is heard.

If the observer is ahead of the cone formed by rotating the generating lines Bd and Be about BS, he hears the following effects:—

(*i*) As the surface *dje* of the sphere passes over him, he hears the high frequency spectrum generated in the supersonic run, which cuts out when the surface *ahb* of the second sphere passes over him. Under present-day circumstances the duration is usually small and he thus hears an isolated bang. This is followed by the sonic pulse generated in the subsonic run after B and also by the sonic pulse generated by the subsonic run previous to O. He does not hear the low frequency spectrum generated in the supersonic run.

If the observer is behind the cone formed by rotating the lines Bd_1 and Be_1 about BS, but ahead of the cone formed by rotating the generating lines Oa_1 and Ob_1 about OS, he hears the following effects:—

(*ii*) As the sections *ad* or *be* of the generating lines of the Mach cone pass over him, he hears both the spectra generated in the supersonic run. These cut out rapidly as the respective spheres pass over him. If the separation of the audio frequencies is sufficient he hears two separate bangs. He probably does not hear the sonic pulse generated after B but only the normal receding Doppler effect. Finally he hears the sonic pulse generated previous to O.

If the observer is behind the cone formed by the generating lines Oa_1 and Ob_1 the following effects are heard:—

(*iii*) As the surface *afb* of the sphere passes over him he hears the lower frequency spectrum generated in the supersonic run which cuts out when the surface *dge* of the second sphere passes over him. This is followed by the normal receding Doppler effect of the motion after *B*. Before all this, of course, he has heard the sonic pulse from the subsonic run previous to *O*.

Another theoretical explanation of the sonic bangs has been offered by other writers according to which the bang is created as a discontinuous process as the accelerating aircraft passes through the velocity of sound, whereas according to the theory offered here the bangs are created continuously by the aircraft above the speed of sound. This difference may be of some considerable importance. If the theory offered here is correct serious problems may arise when two supersonic aircraft travelling in opposite directions pass each other. According to this theory each hears the sonic bang created by the other, probably only one intense bang being heard as the separation of the spectra is small due to the high relative velocity. According to the discontinuity theory this effect does not arise.

It is of interest to note that on this viewpoint the pilot does not hear the sonic bangs as his velocity relative to the source is zero. He hears the continuous noise created by the jets and the aerodynamic sources. The latter increase in intensity in the region of the velocity of sound but this would be continuous. The Doppler effect is not observed in the wind tunnel as the velocity of the observer relative to the source is zero. If the sonic bang is created by a discontinuity in the aerodynamic effects, the pilot, situated as he is in the centre of the disturbance, should hear a tremendous noise effect and further, the sonic bang should be reproducible in the wind tunnel.

The author is indebted to Mr. W. Blackman and Mr. J. A. Booth for instructive critical comment on this work and to the Chief Scientist, Ministry of Supply, for permission to publish this note.

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Measurement of the Profile Drag of Compressor and Turbine Cascades and the Effect of Wakes in Exciting Vibration

by

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THE Melvill Jones equation for the profile drag of a single aerofoil is adapted to the case of an aerofoil in cascade, where the static pressure may be permanently raised (compressor), or lowered (turbine). A simplified procedure for measuring the drag is then described, assuming that the total pressure wake has the form of an error curve. A table of multiplying factors is given, for compressible flow up to an outlet Mach number of 0·9. Many published measurements of cascade drag have ignored this factor, with a consequent error of up to 25 per cent.

The presented method of measuring drag is very rapid, and makes unnecessary the complicated equipment for recording and integrating pressures that is a feature of many cascade wind tunnels.

Finally, a Fourier analysis is applied to a moving row of wakes, to determine the amplitude-frequency spectrum of the downstream total pressure pulsations. It is thought that these pulsations are a likely source of excitation of the subsequent blade rows in compressors and turbines.

1. THE PROFILE DRAG OF A SINGLE BODY IN AN INFINITE STREAM

The force exerted on a solid boundary by a stream of fluid is equal to the rate of loss of fluid momentum in that direction, and Betz proposed, in 1925, the calculation of profile drag from pressure measurements in the wake. But, to quote Prandtl[1]: "Strict proof of the validity of the method is by no means easy..." The practical application of the method owes a great deal to Jones[2] who wrote the drag equation for an aerofoil in the simple form

$$D = \int \rho v_2 (v_1 - v_3)\, dy$$

where ρ is the density, and v_1, v_2, v_3 are the velocities at the three planes shown in Fig. 1: far upstream, just behind the aerofoil, and far downstream. The integral is taken over the wake at plane (2), and the velocities are found in terms of the measured total pressure P and static pressure p at this plane.

As is well known, the introduction of this method marked a big step in the understanding of aerodynamic drag in flight; a modification is now also a valuable tool in the preparation of design data for axial-flow compressors and turbines.

2. THE PROFILE DRAG OF A CASCADE

It is explained in Ref. 3 that the drag component which appears in the calculation of the efficiency of axial-flow compressors is that normal to the line of the cascades. (The friction force along the line of the cascade reduces the work, but not the efficiency). The

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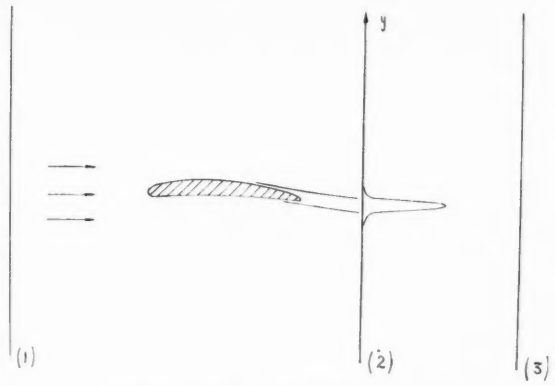

FIGURE 1. Drag of a single aerofoil.

drag can be measured in just the same way as a single aerofoil, by making a total pressure traverse parallel to the cascade across the wakes (Fig. 2). The chief difference between a cascade and a single aerofoil is that the static pressure is permanently changed, because the cascade alters the direction, and hence the velocity, of the fluid. The ability to raise or lower the pressure level of a stream is of course the most valuable property of cascades, and is the basis of the operation of axial-flow compressors and turbines; although cascades are also used in duct corners purely to ensure efficient turning.

Because of the pressure change, the reference plane (3) must have hypothetical properties: the mean flow direction (α_2) of the traverse plane (2), but an unchanged total pressure, $P_3 = P_1$. (*Note*: it does not have the flow that would exist if there were no viscosity, for the downstream flow angle would then be different.)

The drag equation of a cascade blade is thus

$$D = \int_0^s \rho v_2 (v_3 - v_2)\, dy$$

where s is the spacing of the blades. Since the blade chord is usually small, it is easy to space the traverse plane (2) sufficiently far downstream (one half chord or more) so that the static pressure does not vary with y, *i.e.* $p_2 = p_3$. Then, from Bernoulli's equation for incompressible flow,

$$\left(\frac{v_2}{v_3}\right)^2 = \frac{P_2 - p_2}{P_3 - p_3} = \frac{P_2 - p_2}{P_1 - p_2} = g, \text{ say,}$$

and hence

$$\frac{D}{(P_1 - p_2)s} = 2\int_0^1 \sqrt{g}(1 - \sqrt{g})\, d(y/s).$$

This may be rewritten as

$$\frac{D}{(P_1 - p_2)s} = \int_0^1 (1 - g) f(g)\, d(y/s). \qquad (1)$$

Some values of $f(g) = 2\sqrt{g}(1 - \sqrt{g})/(1 - g)$ have been calculated, for the usual range of g (Table I).

TABLE I

$1-g$	0	0·05	0·10	0·15	0·20	0·25	0·30	0·35	0·40
$f(g)$	1	0·987	0·973	0·959	0·944	0·928	0·911	0·893	0·874

From the definition,

$$(1 - g) = \frac{(P_1 - P_2)}{(P_1 - p_2)},$$

so that if $(1 - g)$ is small, and therefore $f(g)$ is close to one, equation (1) can be written approximately as

$$\frac{D}{s} \simeq \int_0^1 (P_1 - P_2)\, d(y/s);$$

that is, the drag is given approximately by the measured mean loss of total pressure. Much work has been done on cascades, with this assumption (*e.g.* Ref. 4), but the error can be quite large.

3. A RAPID METHOD OF MEASURING CASCADE DRAG

Drag measurements on cascades are usually required at a large number of different flow conditions, so it is useful to be able to simplify the procedure. This can easily be done because, unless the boundary layer separates from the blade surface, the shape of the trough of total pressure defect is very close to the error curve

$$\omega = (P_1 - P_2) = \Omega \exp[-y^2/(2\sigma^2)].$$

Ω is the maximum value of ω, and σ is a parameter of the width of the wake. Therefore equation (1) becomes

$$\frac{D}{s} = \Omega \int_0^1 f(g) \exp[-y^2/(2\sigma^2)]\, d(y/s).$$

Unless $f(g)$ is very much less than one, it may be brought outside the integral sign, so that on integration,

$$D = \Omega\phi\sqrt{(2\pi\sigma^2)} \quad . \quad . \quad . \quad (2)$$

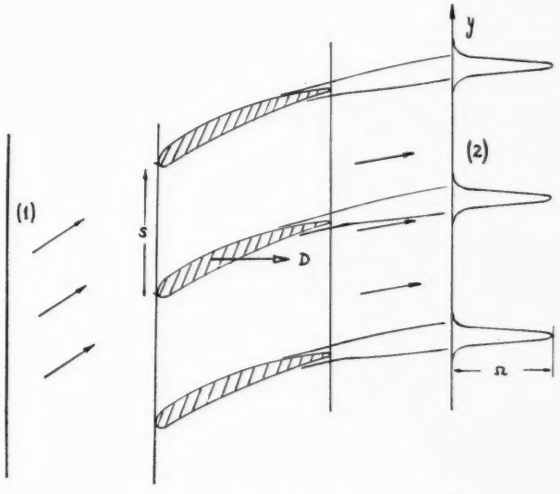

FIGURE 2. The drag of a cascade.

where ϕ is the value of $f(g)$ corresponding to $1/\sqrt{2}$ of the maximum value of $(1-g)$, *i.e.* to $\Omega/[\sqrt{2}(P_1-p_2)]$. Table II can thus be compiled from the definitions of ϕ and $f(g)$.

TABLE II

$\Omega/(P_1-p_2)$	0	0·05	0·10	0·15	0·20	0·25	0·30	0·35	0·40
ϕ	1	0·991	0·982	0·972	0·962	0·951	0·940	0·929	0·918

Instead of σ, it is easier to measure λ, the "width at half-height." Then, from the properties of the error curve,

$$\lambda^2 = 8\sigma^2 \log_e 2$$

and so equation (2) becomes

$$D = 1{\cdot}064\phi\Omega\lambda. \quad . \quad . \quad . \quad (3)$$

The drag coefficient used to derive the efficiency of an axial-flow compressor stage is defined in Ref. 3 as

$$C_D = \frac{D}{\frac{1}{2}\rho v_a^2 s},$$

where v_a is the axial component of the velocity. Hence, substituting from equation (3)

$$C_D = 1{\cdot}064\phi\left(\frac{\Omega}{P_1-p_2}\right)(\lambda/s)\sec^2\alpha_2. \quad . \quad (4)$$

It is essential that both the inside and outside diameters of the traversing tube be small compared with λ. For when a tube points into a stream in which there is a total pressure gradient, the effective pressure on the end of the tube is higher than at the centre of the tube. Thus, if the tube is not very small, λ and therefore C_D will be underestimated. Since cascade blades are often very small themselves, this is quite an important source of error.

Several workers have written papers giving shortened methods for approximate determinations of the drag of *single* aerofoils, by bringing $f(g)$ outside the integral sign in the drag equation equivalent to equation (1). Some, too, have assumed an equation for the wake shape. Lists and discussions of these papers are given in Refs. 6 and 7. In the first paper ever to describe the measurement of the profile drag of *cascades*, Keller[8] used the Betz method of calculation, for incompressible flow, and assumed that the total pressure troughs were triangular in shape.

TABLE III

$\frac{\Omega}{P_1-p_2}=$	0	0·1	0·2	0·3	0·4	0·5
$M_3=0$	1	0·979	0·961	0·938	0·913	0·890
0·2	0·979	0·961	0·944	0·921	0·901	0·879
0·4	0·929	0·918	0·902	0·887	0·868	0·849
0·5	0·895	0·885	0·873	0·859	0·842	0·823
0·6	0·858	0·849	0·839	0·826	0·812	0·796
0·7	0·813	0·810	0·802	0·791	0·780	0·765
0·8	0·767	0·765	0·760	0·752	0·743	0·733
0·9	0·721	0·720	0·719	0·715	0·710	0·702

4. COMPRESSIBLE FLOW

The formulae of the foregoing sections have been derived using Bernoulli's energy equation for an incompressible fluid. The development of the equations for a gas are similar, but more complex[5]. However the drag coefficient of a cascade can still be calculated from equation (4), provided that the value of ϕ is adjusted for the appropriate Mach number M_3. This is, again, a hypothetical value given by the equation

$$\frac{P_1}{p_2} = \left\{1 - \tfrac{1}{2}(\gamma-1)M_3^2\right\}^{\frac{\gamma}{\gamma-1}}.$$

Values of ϕ, as a function of both $\Omega/(P_1-p_2)$ and M_3, calculated from the tables of Ref. 4, are shown in Table III.

Clearly, if the drag coefficient of a cascade is calculated from equation (4) and Table III, the only measurements necessary are of P_1, p_2, Ω, λ and α_2. Thus only three values of total pressures in the wake need be measured, instead of the usual 20 or more; and C_D can be found very rapidly, without the aid of the

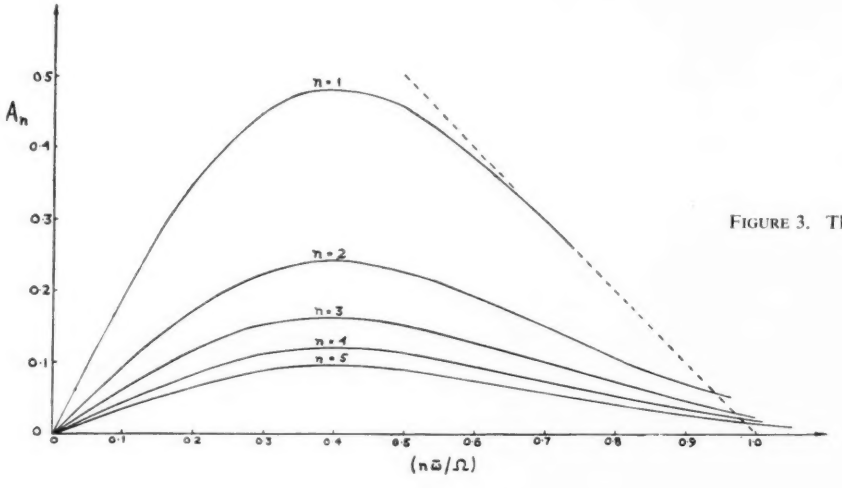

FIGURE 3. The amplitude coefficients.

complicated equipment for recording and integrating pressures that is so often used.

It is clear, too, from Table III, how large is the error introduced by the common assumption that $\phi = 1$.

5. THE EXCITATION CAUSED BY A ROTATING ROW OF WAKES

It has long been thought that the periodic variation of total pressure caused by the wakes of an upstream row of blades is a likely source of excitation of vibrating compressor and turbine blades. It is possible to calculate from the foregoing formulae the amplitude of the different frequency components, by applying a Fourier expansion to a set of cascade wakes. Thus

$$\omega = \bar{\omega} + \Omega \Sigma A_n \cos(2\pi n y/s)$$

where $\bar{\omega}$ is the mean value. For a rotating annular cascade, the distance ratio (y/s) is equal to the time ratio BNt, where B is the number of blades in the row, and N is the rotational frequency. The excitation frequencies at a fixed point are thus BN, $2BN$, $3BN$ ... and so on, and the amplitudes of these components are $A_1\Omega$, $A_2\Omega$, $A_3\Omega$ ... and so on. The coefficients A_n are given by

$$A_n = 4 \int_0^{s/2} \exp[-y^2/(2\sigma^2)] \cos(2\pi n y/s)\, d(y/s). \quad (5)$$

If $(\bar{\omega}/\Omega) > 0{\cdot}7$, A_2, A_3 ... and so on, tend to zero, while A_1 tends to $1 - (\bar{\omega}/\Omega)$; that is, far downstream the curve approaches a pure cosine wave. For $(\bar{\omega}/\Omega)$ less than 0·7, the integral in equation (5) is close in value to the infinite integral, and

$$A_n \simeq 2(\bar{\omega}/\Omega) \exp\{-\pi(n\bar{\omega}/\Omega)^2\}.$$

The numerical values of A_n are plotted in Fig. 3, against $n\bar{\omega}/\Omega$. The curves peak at the abcissa $1/\sqrt{(2\pi)}$.

The dotted line at the end of the curve $n = 1$ shows the amplitude of the cosine wave which the wakes approach, as they mix downstream. The ratio (ω/Ω) increases with the square root of the distance downstream of a point slightly in front of the trailing edges; but the actual value depends on the state of the boundary layer on the blades, and must as a rule be found by experiment for any particular case. The orders of magnitude commonly found are shown in the following example.

TABLE IV

Frequency	*BN*	*2BN*	*3BN*	*4BN*	*5BN*
A_n	0·30	0·23	0·16	0·09	0·04
$\dfrac{A_n\Omega}{P_1 - p_2}$	0·075	0·057	0·040	0·022	0·010

Suppose that, for an unstalled cascade, $\bar{\omega}/(P_1 - p_2)$ is known to be 0·04, and at a distance of one half-chord downstream of the trailing edges $\Omega/(P_1 - p_2) = 0{\cdot}25$. Then $(\bar{\omega}/\Omega) = 0{\cdot}16$, and Table IV can be compiled from Fig. 3. The last row shows the amplitude of excitation as a fraction of the downstream dynamic pressure.

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Dimensions Required for Inspection

by

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1. DISADVANTAGES OF SPECIAL DIMENSIONS

In his technical note on "Dimensions required for Inspection" (July JOURNAL), Mr. G. A. G. Saywell stated that drawings calling up various machined fittings often lack any reference to "practical" measurement by which the operator, toolmaker and inspection department may "set out" and check the work concerned. This statement does not really do justice to the draughtsman who, in preparing the detail drawing for manufacture, has to give attention to all aspects of manufacturing.

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The main purpose of an engineering drawing is to convey technical information in a simple, clear and certain manner to the person or persons engaged on making or inspecting the component. If this condition is to be satisfied it is most important that the dimensions on the drawing be in the form the machine operator requires them, not only as regards style and limited dimensions, but also that the dimensions shall be applied to show the disposition of faces as they will be measured[1]. There should be no more dimensions than are necessary to define the component and no surface, line or point should be located by more than one dimension in any one direction.

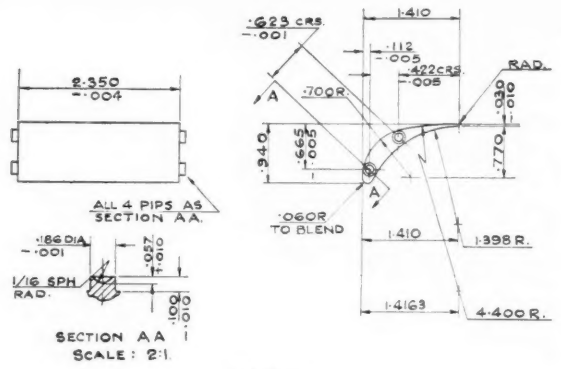

FIGURE 1.

The aim, therefore, of a draughtsman when preparing a detail drawing of a component for manufacture is to bear in mind first and of most importance, the method of manufacture compatible with the design function of the detail. That is to say, the dimensions placed on the drawing, together with the layout of the detail, should be such as to facilitate the manufacture without jeopardising the design function. Bearing these factors in mind, the draughtsman should then consider the subsequent measurement of the component and try to see that, as far as possible, the dimensions are practicable without over-riding his basic consideration. It is no good making detail drawings that are ideal for measurement if they are not suitable for function and manufacture. Mr. Saywell's example does illustrate a problem which has to be faced daily by the operator and inspector, and surely calls for closer collaboration between the planning department and the inspector. The planning layout should give any additional dimensions that may be required to facilitate inspection and these dimensions should be carefully checked before the layout is issued to the shops. It is quite true, as Mr. Saywell states, that most of the essential calculations will be made by the draughtsman at some stage and reference could be made to the drawing office with regard to any special dimension that the planning department requires. It is not reasonable, however, to put on the drawing special sets of dimensions to facilitate "direct" measurement. It is permissible to use reference dimensions where such dimensions are not strictly necessary to define finished products but provide useful information. Such dimensions must be clearly marked "Ref."

A good case of such dimensioning is with gear centres where it is normal to give them by ordinates and then to provide a direct dimension between centres, which is marked as a "reference," and is itself most useful for inspection purposes. Production processes or inspection methods should not be specified unless they are essential to ensure satisfactory functioning or interchangeability. This does not apply to process drawings. It is normally assumed that the function of providing inspection gauges and so on, and therefore, deciding the method of inspection is the prerogative of the planning department. Different methods of

FIGURE 2(*b*).

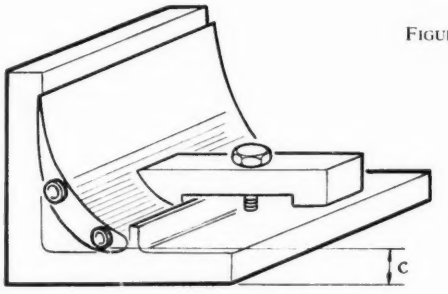

FIGURE 2(*a*).

inspection will be laid down, (*a*) for small quantity production for development, (*b*) for large scale production.

An example of such varying methods of inspection is illustrated in connection with Fig. 1, which is a deflector vane from the Ghost engine diffuser casing and concerns the position of the pips relative to the vane form, and in particular to the 0·112 in. −0·005 in. dimension, being the centre of one pip relative to the vane form.

For small quantities a method of inspection for this dimension is shown in Fig. 2(*a*) and is a simple fixture whereby the vane is located on horizontal and vertical planes relative to similar planes on the drawing. It being impossible to measure to the centre of the pip after manufacture, the inspection dimension is taken from the underside of the pip to the edge of the vane form, that is to say the 0·112 in. −0·005 in. dimension less half the diameter of the pip; it will be seen from Fig. 2(*a*) that the actual measurement (dimension *c*) is taken from the underside of the pip to the surface table and thereby takes into account the base thickness of the fixture.

The method of measurement for large production quantities is illustrated in Figs. 3, 4 and 5, and it is a most interesting measuring gauge. Figure 3 shows the basic fixture with the measuring clock attachment set slightly back. It will be noticed that on the fixture there are two steps *A* and *B* which locate the pips on the deflector vane when fitted in the fixture as shown in Fig. 4. Figure 2(*b*) shows the dimension that is measured on this gauge; it is in effect a dimension from the vane form to the farther side of the pip, *i.e.* 0·112 in. −0·005 in. plus half the diameter of the pip, and the dimension from step *A* to the surface *C* shown in Fig. 3 is the required dimension. When the blade is set in position, the measuring clock is moved forward until the clock on one side, that is to say the side being measured, is set at a nominal position on the gauging face *C* of the fixture. Then the clock is moved sideways on to the vane form as shown in Fig. 5, and the variation read on the clock, which records in

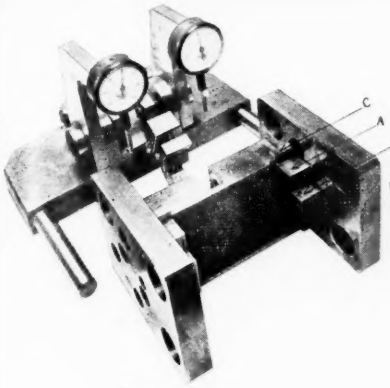

FIGURE 3.

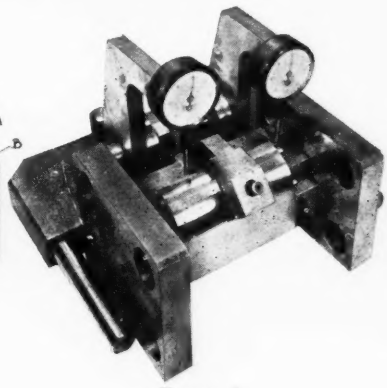

FIGURE 4.

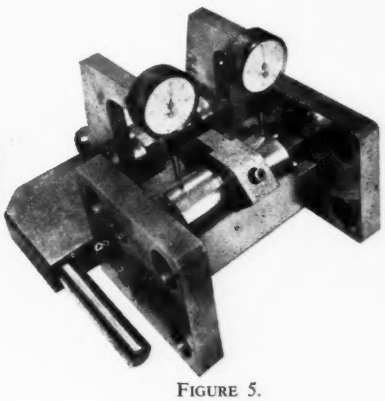

FIGURE 5.

thousandths of an inch, shows the error of pip position relative to the vane form.

It will be seen from these illustrations that whereas, for manufacture, the positioning of the pips relative to the vane form must of necessity be related to the centres of the pips, when it comes to inspection the inspector must rely on the use of the outside diameter of the pip as a measuring datum. If it became necessary to cater for inspection dimensions on the detail drawing in the case illustrated, immediately two further sets of dimensions would have to be applied to the drawing. This is not practicable and the only sound and correct method on any detail drawing is to dimension first and foremost for ease of manufacture relative to functional surfaces, bearing in mind inspection necessities, but not to put on the drawing inspectional dimensions.

It is interesting to note here that this fixture, by resting on each of its four sides in turn, is used to measure the position of the pips relative to the form in each direction.

2. FUNCTIONING, MANUFACTURING AND GAUGING DATUMS

Where two or more components operate together to form a functioning unit, it is important that the functioning surfaces on the individual components are chosen as datums and the basic dimensions are given directly from such surfaces[2]. The selected basic dimensions should achieve the desired function of the assembly, with the minimum number of dimensions. In good detail design, functioning, machining and gauging datums will in general be identical and the detail drawing should be so arranged to achieve its objective.

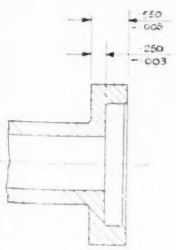

FIGURE 6.

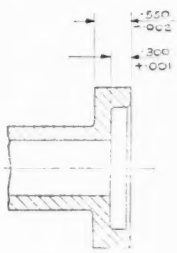

FIGURE 7.

If datums other than function surfaces are chosen then the tolerance for dimensions will inevitably be required to be reduced. It follows, therefore, that tolerances should be chosen with due regard to problems and possible defects in manufacture. Inspection and assembly tolerances should in general be as large as satisfactory functioning will permit. A simple but typical case of tolerances (*a*) in association with functioning and (*b*) modified with due regard to manufacture and inspection is shown in Figs. 6 and 7.

Figure 6 shows the ideal condition that would be required in the case where it is necessary to associate the bottom of the large counterbore with the underside of the outside flange. This, of course, is not a good thing for manufacturing or inspection but does permit a fairly wide tolerance of 0·005 in. on the dimension controlling the width of the flange and a reasonable tolerance of 0·003 in. for the dimension required from the bottom of the counterbore to the underside of the outside flange. If, however, this component is re-dimensioned to facilitate manufacture and inspection, then it is necessary to reduce these tolerances to keep within the ultimate assembly limits, and Fig. 7 shows the result whereby there is a 0·002 in. tolerance on the thickness of the flange and a 0·001 in. tolerance to the bottom of the counterbore.

During and since the end of the last war an increasing amount of thought has been given in the drawing office to production methods and the subsequent inspection of finished components. This has been made necessary as a result of the introduction of less skilled labour. One most important way in which progress has been made and as a result of which help has been given to the inspection department, is on the control of positional tolerances, particularly in reference to diameters and their relation to concentricity and squareness.

Previously the standard practice had been merely to add a note on the drawing to the effect that "limited diameters" were to be concentric with one another and square with shoulders and faces, etc., to within, say, 0·001 in. total clock reading. This has now been modified and far more closely controlled by the introduction of the proposals given in C. A. Gladman's paper[2] and illustrated in Figs. 8 and 9, which are drawings of a proposed extension shaft.

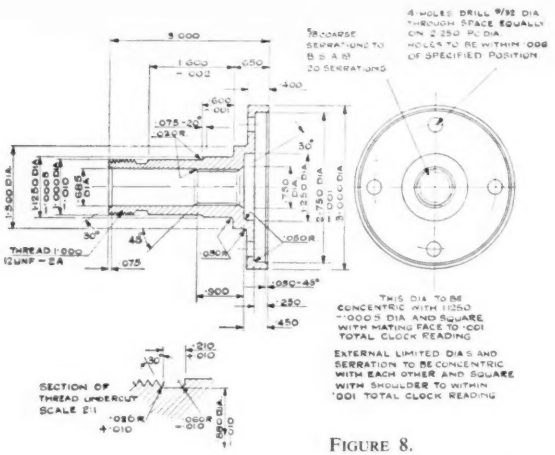
4-HOLES DRILL 9/32 DIA THROUGH SPACE EQUALLY ON 2·250 PC DIA
5/8 COARSE SERRATION TO B.S. A 19 20 SERRATIONS
HOLES TO BE WITHIN ·006 OF SPECIFIED POSITION
THREAD 1·000 12 UNF - 2A
THIS DIA TO BE CONCENTRIC WITH 1·1250 ·1000 5 DIA AND SQUARE WITH MATING FACE TO ·001 TOTAL CLOCK READING
EXTERNAL LIMITED DIA S AND SERRATION TO BE CONCENTRIC WITH EACH OTHER AND SQUARE WITH SHOULDER TO WITHIN ·001 TOTAL CLOCK READING
SECTION OF THREAD UNDERCUT SCALE 2/1

FIGURE 8.

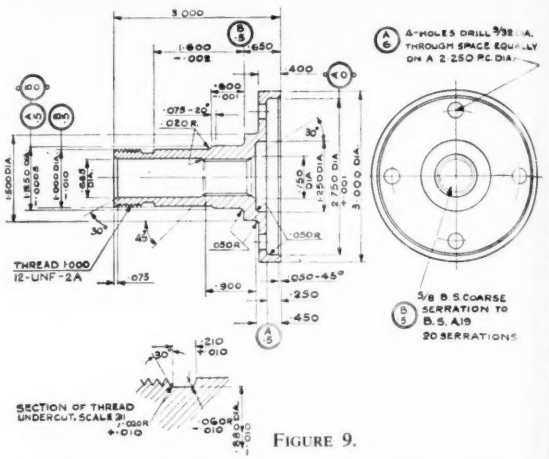
4-HOLES DRILL 9/32 DIA. THROUGH SPACE EQUALLY ON A 2·250 PC. DIA.
5/8 B.S COARSE SERRATION TO B.S. A19 20 SERRATIONS
THREAD 1·000 12-UNF-2A
SECTION OF THREAD UNDERCUT, SCALE 2/1

FIGURE 9.

Figure 8 shows the standard previously worked to whereby the concentricities and squareness were covered by a note. The difficulty in interpretation of this drawing with respect to these points by the operator and inspector can readily be seen. Fig. 9 shows the new presentation of this type of component and involves the use of two symbols, the shape and meaning of which are as follows:

(*a*) *The Datum Symbol*

This is a datum symbol, used to denote a datum surface. The letter of the datum, which may be *A B C D* and so on, indicates the group to which all other symbols in the same group are related. The figure "0" indicates that the tolerance is nil; in other words the surface is made a master datum.

(*b*) *The Group Symbol*

This is a group symbol, the letter indicating the group to which it belongs. The figures 1·2 are expressed in units of 0·001 in. and denote that the surface thus marked shall be within 0·0012 in. of its specified position, that is of its theoretical position relative to the datum surface.

With such a component it is necessary to select carefully the most important functioning dimension and set it as the basic datum. Here, in Fig. 9 the 2·750 in. + 0·001 in. diameter, which spigots on to its mating part, has been chosen and is given the symbol A 0

Other limited diameters and faces associated with this spigot are then noted in association with it, *e.g.* the bearing diameter 1·125 in. − 0·0005 in. is noted A ·5

This bearing diameter in itself is also a main functional dimension so it is given the additional setting datum of B 0 and the other limited diameters and shoulders associated with it can be controlled accordingly, *e.g.* the $\frac{5}{8}$ in. serrations have been noted B ·5

thus indicating that this diameter must be concentric with B 0 to within 0·001 in. total clock reading.

Figure 9 also illustrates how these symbols can be used for indicating limits for position. The four $\frac{9}{32}$ in. clearance holes are required to be within 0·006 in. of their specified position in relation to the 2·75 in. + 0·001 in. diameter spigot. The symbol A 6 thus indicates this fact and means that the axis of each clearance hole must be within 0·006 in. of its geometrical position, *i.e.* within a circle of 0·006 in. radius, the centre of which is the theoretical position. This system of symbols is also readily applicable to limits for symmetrical work.

CONCLUSIONS

On the whole far more attention is paid today by the drawing office to all aspects of manufacturing. Inspection requirements are carefully considered and taken into account when dimensioning manufacturing drawings although, as previously stated, it is not practicable to place on the drawing special dimensions for the inspector.

ACKNOWLEDGMENT

The author acknowledges the permission of The de Havilland Engine Company Limited to publish certain information given in this note. Any opinions expressed are the author's own and not necessarily the considered opinion of The de Havilland Engine Co.

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Reviews

SOUND BARRIER. *Neville Duke and Edward Lanchbery.* 117 *pp. Diagrams and photographs. Cassell,* 1953. 8*s.* 6*d.*

This is not "the book of the film" nor does it deal exclusively with the problems of supersonic flight; it tells "the story of high-speed flight for the general, non-technical reader."

The explanation of aeronautics to the large body of interested but non-technical readers, ranging from schoolboys to cabinet ministers, is a task worthy of great efforts. A few scientists have succeeded (Sutton's Pelican "The Science of Flight" is a good example) and now one of our greatest test pilots makes his contribution.

The book starts well by explaining that the sound barrier is not a barrier at all, but the pace is so great that within six pages the reader is studying the expansion and shock waves associated with the supersonic flow over an aerofoil, transonic wind tunnels and the Lockheed "Bump." By only the fourteenth page the various theories of the sonic bang are under discussion. Then the development of engines and of airframes to achieve high speeds and the physiological limitations of the pilot concerning pressure, temperature and acceleration are discussed. The authors should have taken more care not to undermine the reader's confidence by careless use of technical words; the sixth form boy may well understand the meanings of "momentum" (p. 6), "fluid" (p. 11) and "camber" (p. 52 and 54) and not agree with the authors' uses of these words. The reader who has already learned some aerodynamics will not agree that skin friction has little effect at low speeds and that drag is primarily form drag (p. 48) nor that the high nose-up attitude of the Delta at landing increases the chances of a stall (p. 58).

The account of the pilot's physiological problems is very well given although it is surprising to read that a pilot "would not be able to tolerate a temperature of 116°C. beyond half an hour." (Hence the term "hard boiled," perhaps.) In the section on ejector seats we learn that the pilot must not be subjected to more than $18\frac{1}{2}$ *g* and that "the most important fact is that the rate of *g* shall not exceed 250 to 300 *g* per second." Even the scientist accustomed to triple differentiations will find it difficult to appreciate the physical significance of this limit.

The concluding comments are on current aircraft designs and "things to come." The Contents page lists the heading "The speed of light and its implications" but no reference to this could be found in the text. Has Security concealed a new Hawker development from us? Will Mr. Duke's "sonic boom" be followed by the "optic flash"?—A. H. YATES.

HUMAN FACTORS IN AIR TRANSPORTATION. *Ross D. McFarland, Ph.D.* 830 *pp. Illustrations. McGraw-Hill, New York,* 1953. 106*s.* 6*d. net.*

This is a very ambitious work in which the author has compiled and united a wide variety of specialities and sciences relating to the human aspects of air transport. The aim has been to make it a textbook for students and, at the same time, a reference volume for doctors and executives engaged in both civil and Service aviation. Unfortunately, although the book contains a wealth of facts and statistics as well as an extensive bibliography of 1,213 references, so much unimportant detail is included that the student will find it difficult to garner the facts. There are 155 illustrations which are generally good, and many of the 170 tables provide excellent statistics in a readily assimilable form, which should prove valuable to airline medical officers and executives.

There are 17 chapters, dealing respectively with: the nature and extent of airline operations; psychological factors in the selection of airmen; the role of physical examination in selection; indoctrination and training of flight crews; periodic health examinations for airmen; maintenance of physical and psychological fitness in air crews; operational aspects of fatigue; ageing and efficiency in air crew; selection and placement of ground staff; maintenance of health of ground staff; prevention of industrial accidents in ground operations; survival and rescue in aircraft accidents; sanitary control of airports and ground facilities; health and quarantine in air travel; care and contentment of passengers; transport of patients by air; and the organisation and functions of air transport medical programmes.

Naturally, in a book of this nature, there is little new as its main purport is to be factual. The views expressed are largely the impressions of the author as the result of his extensive experience in air transport and industrial hygiene. At the same time much has been culled from the voluminous literature on aviation medicine and allied subjects as well as from proceedings of Conferences, Congresses and Meetings in which the author has taken part. This held he generously acknowledges in the preface.

The book gives a good picture of the North American conception and practice of aviation medicine in relation to Commercial flying, which is not entirely in accordance with those on this side of the Atlantic.—H. E. WHITTINGHAM.

HEAT TRANSFER PHENOMENA. *R. C. L. Bosworth.* 211 *pp. Associated General Publications, Sydney,* 1952. 42*s. net.*

This book is more general than its title suggests. It is really an essay in the propagation of a property by a "carrier" (to use the author's terminology). The author is, of course, primarily concerned with the propagation of heat regarded as a particular case of this method. But he is at pains to keep before the reader at all times that the technique is also applicable to mass transport and to electrical conduction.

The book is an expanded version, in monograph form, of the introductory lecture delivered to a Conference at Sydney. It is not possible to be certain which is the added material, but this reviewer thinks that a reasonably good guess can be made. If his guesses are correct then it is doubtful if the expansion has been a real improvement. The balance between the various aspects, Conduction, Convection and Radiation, of Heat Transfer is uneven; in general a designer would search in vain for detailed formulae and it assumes rather a lot of general background knowledge for a student. In short it is not easy to specify to whom this book is addressed.

On the other hand anyone with some previous knowledge of the subject of Heat Transfer will benefit from a book of this type with its insistence on the close analytical resemblance that exists between heat and mass transfer or between the flow of electricity and of heat. The list of references is unusually complete and the book has a good index both of authors and of subjects. It seems most likely that research workers will benefit most from this book and it may be that the author had them in mind when expanding his original paper.—W. F. COPE.

THE SCIENCE OF PETROLEUM. *Vol. V. Part II. Joint Editors, B. T. Brooks and A. E. Dunstan. Oxford University Press, London 1953. pp. 329. Illustrated. Price 100s. net.*

The original four volumes of the *Science of Petroleum* were published in 1938, and it was stated that supplementary volumes would be published, when necessary, to bring the subject matter up to date. In accordance with this plan Vol. V was to bring up to date the subject matter of the Chemistry, Physics and Chemical Engineering and Part I of this volume appeared in 1950 covering the literature up to 1946 on the composition and properties of crude oils. The present book is Part II of this volume and deals with Synthetic Products and Refinery Processes. The Editors state that Part III of Volume V is to be published bringing Parts I and II up to date. This is essential if the original plan is to be maintained, for the present text, like Part I, covers the literature up to 1946.

Section I, Synthetic Products, contains seven articles. Section II, Refinery Processes, is divided into five parts, (1) Distillation, (2) Refinery Processes, (3) Cracking, (4) Hydrogenation and Dehydrogenation, (5) Alkylation.

It is essential to realise that the articles were completed in 1946, and that with one or two exceptions the authors have not had the opportunity of bringing the subject matter up-to-date. This presumably will be done in Part III which it is expected will also include articles on new processes such as Fluid Hydroforming and Platforming.

The present text gives an excellent review of developments in the Chemistry and Physics of Petroleum Processing during the period 1936-1946, a period in which many new processes were developed and in which the catalytic era may be said to have been born. For this reason the chapters on Catalytic Cracking are of particular importance and will remain the standard reference texts on the early development of these processes for many years. The chapters on Distillation are equally important contributions and contain much valuable material. Their value would, however, have been increased had they included details of tray design, bubble cap plate hydraulics, some account of development in packed tower design and methods of calculating furnace design. It is hoped that Part III will cover these aspects.

The articles in Section I which deal with the utilisation of olefines for polymers and chemicals were written at the beginning of the expansion of the Petroleum Chemical Industry. Much of the information is still of value, particularly that on the properties of butadiene, styrene and synthetic rubbers. The pattern of industrial development is, however, now much clearer and several of the processes mentioned have not achieved the commercial significance originally anticipated. Since 1946 a great deal has been accomplished and much new information has become available, making it difficult to assess the value of these particular chapters by reference to the position at the end of 1946. Similarly the sections on hydrogenation and alkylation suffer by comparison with new developments. The hydroforming process for instance has been revolutionised by the application of fluidised technique while new hydrogenation-dehydrogenation plus cracking catalysts have been developed into commercial operations such as Platforming. In 1946 theories concerning the mechanism of alkylation were much discussed, since then the carbonium ion mechanism proposed by G. Schmerling has been widely accepted. The process has not the same vital importance in peacetime that it had during 1940-45 and recent developments have been concerned essentially with reactor design and increased acid utilisation, factors which were only beginning to be studied in 1946.

These comparisons should not be allowed to detract from the merit of the text, which as a record of the developments during 1936-1946 is of the greatest value. The book is produced with the same care as previous volumes, is amply illustrated and indexed but the Editors will be well advised to ensure that Part III of Volume V is published with as little delay as possible.—F. MORTON.

GRUNDLAGEN DER AEROMECHANIK UND FLUGMECHANIK. *Arthur Pröll. Springer. Vienna 1951. 612 pp. 278 figures. 82s. net.*

Professor Pröll held the Chair of Aeronautical Engineering at the Technical University of Hanover for more than a quarter of a century. He contributed greatly to the evolution of the modern sailplane and was also well connected with aero-mechanical developments and with basic research. In 1919, he published a textbook on aero-mechanics which was well received. The present book is an expanded and revised re-edition; the reviewer approached it, therefore, with high expectations.

Although the book has grown considerably, it can hardly be said that the thirty odd years of aeronautical development which have passed since the "Flugtechnik" was published, are duly represented; the book is too much an improvised re-hash of the 1919 edition to form a suitable textbook for a modern student. It is too much dated, and many subjects of great importance today are dealt with perfunctorily. Entire chapters are verbally reprinted; this explains, for example, why the reader is referred to Eiffel's pre-1914 experiments on drag and scale effect, and why the upper atmosphere is ignored (in spite of reference to an informed American paper of 1947).

Jet propulsion is dealt with on a page or two only, six lines are devoted to the ram-jet (explaining it as an "open rocket"); while rocket propulsion gets away with three lines. Laminar flow aerofoils occupy less than two pages, boundary-layer manipulation and slotted wings receive about the same space, and if there is something about flaps, the reviewer must have overlooked it. Variable-pitch airscrews are mentioned as a practical possibility (three-quarters of a page), but 20 pages are devoted to a broad explanation of the elements of the (older) airscrew theory, with omission of the more recent theoretical developments.

The stability treatment is mostly reprinted from 1919. Although the conception of the aerodynamic centre is dealt with in connection with the classical methods of the conformal transformation of incompressible flow, no use of this is made for a simplified treatment of stability. Not even the elementary remedies for dealing with eccentric thrust forces are mentioned, and should a student follow the recipes which are given for equilibrium in pitch, he ought to finish with a centre-of-gravity position which would prove too unhealthily far aft.

Unfortunately, too, not much care has been taken in the editing of this book (which is uncommon with a Springer publication). Occasionally, a figure is placed upside down (Fig. 234), and there is a lack of consistency in the notation which is very confusing.

Admittedly, the treatment of the fundaments of hydrodynamics and gas dynamics (32 pp.) is good and clear. But as an aeronautical textbook, the scope is too restricted while many of the basic parts could be appreciably shortened. Stratospheric flight, sailplanes, parachutes, helicopters and gas turbines are not mentioned at all and work done outside Germany is disregarded.—A. R. WEYL.

THE MOLECULAR THEORY OF FLUIDS. *H. S. Green. 264 pp. Diagrams. North-Holland Publishing Company, Amsterdam. 20s. net.*

The appearance of a systematic exposition of the molecular theory of fluids by one of the leading authorities on the subject is an event of great importance, both for those who are doing research in this field and for the more general reader who wishes to acquaint himself with the modern kinetic theory. This book will fill a gap which has existed now for some years in the literature of this important branch of physics. Professor Green adopts at the outset the powerful, descriptive apparatus of distribution theory, and uses this apparatus clearly and elegantly throughout.

Chapter I presents a kinetic picture of the molecules of gases and liquids and relates the macroscopic properties of fluids in a qualitative manner to the collisional processes by which they exchange their energy and momentum.

In Chapter II, which is an introduction to statistical thermodynamics, he derives Boltzmann's theorem in an elementary manner which avoids the lack of rigour so frequently associated with so-called proofs of this theorem, and establishes the fundamental connections between the molecular distribution functions and the thermodynamic properties.

Chapter III, on fluids at rest, is concerned with the experimental and theoretical determination of the so-called radial distribution function, and in this Chapter the author describes the superposition approximation on which he and Professor Born based their well-known theory of liquids. Professor Green, however, has taken the trouble to discuss the very recent modifications of the Born-Green theory, in particular that of McLellan who has solved the usual non-linear integral equation without attempting to linearise it first. Perhaps the weakest section in the book is the end of this Chapter in which the author is discussing intermolecular forces. His use of the expression "associated liquid" does not accord with ordinary usage, and this might create confusion in the mind of a physical chemist acquainting himself with the modern kinetic theory. For example, the sentence "Therefore the only type of molecule which can form an associated liquid is one in which the electrons are not already associated in pairs so that there is a resultant electronic spin" would rule out water and hydrogen fluoride from the class of associated liquids, which the physical chemist would be reluctant to do.

In Chapter IV a full discussion is given of the nature of phase transitions. The exposition of Mayer's condensation theory is the clearest that the reviewer has seen anywhere, and the assumptions underlying Mayer's theory are brought out with admirable clarity. The discussion on freezing is necessarily more sketchy in view of the smaller amount of work which has been done on this topic. The Lennard-Jones-Devonshire cell model of a liquid is consigned to the end of the Chapter, representing as it does a more restrictive set of assumptions than those favoured by Professor Green.

Chapters V and VI are devoted to transport properties. The relations obtained there are formal rather than practical and in so far as the results depend on the expansion of the distribution function about its Maxwellian value they will not be expected to apply with any accuracy to the extremely high gradients associated, for example, with shock waves.

Chapters VII and VIII complete the classical theory of fluids. In the former the elasticity of a liquid, its surface properties, dielectric properties and the problem of Brownian motion are discussed. In the latter the methods of Enskog, Chapman and Cowling are used to derive exact formulae for coefficients of viscosity and thermal conduction, and the molecular mechanisms of transport in gases and in liquids are differentiated clearly.

The last Chapter, Chapter IX, is an introduction to the quantum theory of fluids. This shows how the classical equations of motion must be modified when Planck's constant cannot be neglected, and how the nature of the statistics obeyed by the molecules introduces an important modification in the density matrix at low temperatures. Liquid helium is the only material for which quantum effects are particularly important, so much of the discussion centres on it.—H. C. LONGUET-HIGGINS (King's College, London).

PROCEEDINGS OF THE SOCIETY FOR EXPERIMENTAL STRESS ANALYSIS. *Volume X, Number 2. Edited by C. V. Mahlmann and W. M. Murray. Published by the Society for Experimental Stress Analysis, Cambridge, Massachusetts, 1953. 176 pp. Diagrams, photographs, index.*

Apart from two, all the papers in the present volume are of direct interest to aeronautical engineers.

There are three papers on the measurement of stress under conditions the reverse of straightforward. One tackles the problem of measuring the stress in a metal sheet—the stressed-skin of a wing for example—when only one side of the sheet is accessible, and where the resultant stress gives only the sum of the bending and direct tensile stresses. The method described relies on incorporating in the "strain-rosette" a means for measuring the change of curvature as well as the resultant stress. The second of the trio describes the use of stress-coat technique for measuring the centrifugal stresses in a turbo-supercharger impeller. It depends on using a calibrated rotor that spins with the impeller and is therefore subjected to precisely the same physical conditions. In the third paper it is shown how the method of optical interference may be used to obtain reliable quantitative (as distinct from qualitative) estimates of residual stresses.

The problem of structural fatigue today looms large on the aeronautical engineer's horizon and four out of the set of five papers on this subject have a direct interest for him, the odd paper having to do with fatigue failures in road-transport trailers. One paper deals with the stress-life of helical gear teeth and the various factors that govern it. The value of the paper is much enhanced by some excellent photographs of typical failures. Another paper in the set describes a method of measuring the increase in the damping of turbine blades as a result of deliberately mounting them loose in their sockets. A valuable paper from the National Bureau of Standards describes a new axial-load fatigue-testing machine with the novel feature of "dual amplitude." The last paper of the quintette gives the results of experiments to determine the effect of the superposition of stress-raisers—to find, in other words, the effect of a notch within a notch.

Of the two papers concerned with structural vibration, one describes the use of a tuned mass (of negligible amount compared with the mass of the structure investigated) for successively measuring the vibration amplitudes at different points on a vibrating structure. In the second it is proposed to determine the constraints acting on a bar or ring when forming part of a structure by carrying out various resonance tests on that structure. To be frank, the present reviewer found it difficult to understand; readers of the book may decide who is to blame.

A most valuable "Cumulative Index" of all the papers so far published by the Society forms an appendix to the present volume.—D. WILLIAMS.

AERODYNAMICS OF PROPULSION. *D. Küchemann and J. Weber. McGraw-Hill, London* 1953. 340 *pp. Diagrams.* 64*s.* 6*d. net.*

The very general title of this book may well mislead the reader into expecting more than the authors intend. To quote their preface, it is "an account of some of the work done in this field, and problems that have arisen . . . it cannot be expected to be exhaustive, and it necessarily reflects the authors' preferences." Generally the treatment is theoretical (although mathematics is kept to a minimum), and practical problems are not discussed in detail, the aim being to provide an understanding of basic processes and flow phenomena.

After a short introductory Chapter, Chapters 2 to 5 cover generalities, respectively the fundamental thermodynamics of propulsion processes, the general theory of two-dimensional non-homogenous flow, the theory of design of air intakes in potential flow, and the theory of finite-length fairings (including ring aerofoil) in potential flow. Chapters 6, 7 and 8 discuss in turn the ducted propeller, the ram-jet, and the turbo-jet engine. Chapter 9 deals with installation problems, mainly concerned with inlet flow losses and external drag of turbo-jet engines. Chapter 10 concerns the effect of the jet efflux on adjacent surfaces. Chapter 11 discusses animal flight and the potentialities of more direct mechanical imitation. Chapter 12 deals with problems relating to the installation of coolers of various types. There is an Appendix, with tables, giving values of the stream function and velocity components for such singularities as vortex and source rings, used in the theory of intake design.

The authors exclude from discussion the design of propellers, turbo-machinery, and reciprocating engines, on the grounds that adequate specialist texts are readily available, and this seems well justified. There are minor omissions, however, for which the same cannot be said: it is surprising to find no mention of the special problems of air intakes behind propellers, or indeed of propeller slipstream effects in general. Again, it is hardly adequate to refer the reader to Schairer's paper given at the Third Anglo-American Aeronautical Conference (1951) for information on jet nozzle problems; although this is a useful paper it does not pretend to be comprehensive, and no references are given in it to lead the student further.

Problems of supersonic flow are not treated, except for a brief and, perhaps inevitably, unsatisfactory section in the chapter on ram jets.

It is clear that the "authors' preferences" are for subjects which can be exactly treated by mathematical theory, and Chapters 2 to 7 are correspondingly good, and should be illuminating to the student, although certain criticisms could be made. This reviewer does not feel, for example, that the use of different symbols, with opposite sign conventions, to denote work done on and by the working fluid really makes for clarity. But when the subject does not permit exact mathematical treatment, as in most of the subsequent chapters, the standard definitely falls. Much useful information still appears, but the sections in which the authors were not really interested are too apparent by their perfunctory nature. Even allowing that the authors do not intend to provide "working formulæ," the discussion often seems insufficiently grounded in actuality—for example, there is a discussion of the relative effect of inlet and exhaust pipe losses on turbo-jet thrust which is based on the assumption of constant volume flow operation, which is unlikely to occur in practice. A general point of criticism throughout the book is the lack of qualifying phrases: for example, the reader will be left with the impression that any thermometer exposed in an air stream indicates stagnation temperature (p. 26); that a free streamline can always be replaced by a solid wall without affecting the flow pattern, even in viscous flow with an adverse pressure gradient (p. 39); that diffuser losses depend only on area ratio (p. 95). It is irritating, too, to be told twice that losses in static thrust due to air intake lip breakaway can easily be avoided by increasing the lip radius, when it frequently happens that there is insufficient room to do so. The discussion of air-air intercoolers on pp. 295-7 appears to be based on incorrect equations.

There is much of value in this book, but it would be greatly improved by a thorough revision of such minor points. The price is high for a book which covers an arbitrarily restricted part of its field, and the reviewer would class it as one which libraries should possess, and students should read.—W. J. D. ANNAND (Rolls-Royce Ltd.).

AERONAUTICAL ENGINEERING. VOLUME II. *R. Tatham. English Universities Press, London 1952. 215 pp. Illustrated. 21s. net.*

Volume II of "Aeronautical Engineering" extends the structural section of Volume I published in 1949 by the same author. Both Volumes form part of the Technical College series which is now under the general editorial eye of Dr. W. E. Fisher.

Mr. Tatham has catered for the students preparing for the A1 and A2 courses of the Higher National Certificate. In view of the author's background as a lecturer at Loughborough and as a stress analyst in the Industry, it is hardly surprising that he has been able to blend the academic treatment with the strictly practical aspects of the subject in a most satisfactory manner. A particularly desirable feature of the book in this respect is a concise summary at the head of each chapter explaining why the work of that chapter is necessary, its applications and the principal limitations.

The treatment of the work is conventional and should meet with the approval of most lecturers requiring a suitable textbook on which to base their A1 and A2 structural courses.

Two chapters are of particular note. In one, the author gives an introduction to the general nature of the requirements of A.P. 970 and B.C.A.R. which should ensure that the student does not finish his aircraft structures course in complete ignorance of the fact that aircraft have to be built to conform to the requirements of the authorities in these matters! In the other, which is one of three appended chapters, an introduction is given to the Hardy Cross method of moment distribution. The reviewer feels that it is unfortunate that Mr. Tatham did not see fit to do more than mention the reduction in labour brought about in the balancing process by modification of the stiffnesses to allow for simple supports. It is not possible to appreciate the full scope of the method of Hardy Cross until this modification has been demonstrated.

The book is profusely illustrated although the poor quality of a number of the diagrams detracts from an otherwise neat presentation. Volume I of this book was criticised on similar grounds in a review in this Journal in 1950.

Higher National Certificate aeronautical students will be well served by this Volume and in the absence of a modern

British textbook on aircraft structures for University students, it might well form a useful addition to the reading matter of the prospective B.Sc. (Eng.) taking aeronautical subjects.—A. J. BARRETT.

AIR TRANSPORT IN INDIA (GROWTH AND PROBLEMS). *M. R. Dhekney. Vora & Company, Bombay,* 1953. 248 *pp.* 6 *Rupees* 8 *Annas net.*

The Indian Government has recently passed a Bill nationalising the country's Air Transport industry. The 14 existing companies, some privately owned and some with Government capital, are being merged into one International and one Domestic Corporation. Thus the time is most opportune for a book such as this, which describes in terms suitable for the general reader, the requirements for safety, flexibility, frequency and so on that govern any airline; which discusses in considerable detail the traffic potential and the aircraft available in India, and the histories and the economics of the airline companies; which examines the problems of finance, price fixing, personnel, organisation and licensing that have grown up since the war, and finally relates them to the Government organisations of ground facilities and Government expenditure.

Mr. Dhekney is an economist who served on the Indian Air Transport Enquiry Committee of 1950, and he has produced a book which, in spite of a little wooliness in some of the arguments and a certain amount of repetition, is pleasantly written and easy to read. It contains a map of the Indian Air Routes and numerous tables of figures which illustrate every section of the book. It thus forms a most useful introduction to airline problems generally for those interested in obtaining a broad picture of this kind, but for students and others particularly interested in Indian aviation it will form an invaluable guide to the conditions that led up to the present nationalisation proposals. Except for the very large number of printing errors, which is by no means covered by a page of errata at the end of the book, and the rather poor standard of printing, it is very good value at its price of just under 10 shillings.—H. B. CUNDALL (Air Registration Board).

I FLEW FOR THE FUHRER. *Heinz Knoke. Translated by John Ewing. Evans Brothers, London,* 1953. 187 *pp. Illustrated.* 12*s.* 6*d. net.*

When the young men of the R.A.F. hastily zipped flying suit and tightened Mae West as they ran to their waiting Hurricanes and Spitfires at the warning of "scramble," one's heart seemed to go with them in mingled pride and apprehension at their battle against Nazi airmen inevitably imagined as darkly satanic and discreditable. Yet even if the pages of Knoke's book are turned but casually it is easy to see that the spirit of the *Luftwaffe* was almost exactly the counterpart of the R.A.F., both well served by gallant youth. True the German seemed to express his patriotism as though it was dogma drilled into him rather than an upwelling of the heart, but it was none the less intense, and his fears and hopes, thought and action were identical with any other man who makes the air his medium.

Thus when Knoke crashes in flames, only to roll from the debris and find beside him the American pilot he has just shot down, he writes of the incident: "We have a friendly chat for half an hour. He seems like a decent fellow. There is no suggestion of hatred between us, nor any reason for it. We are both pilots, and we have both narrowly escaped death."

Written in terse diary form covering the War years, the book, judging by its uniform phrasing, may be the compilation of a later period, in the light of which comment has been judiciously inserted on such things as failure of the Nazi regime, war crimes and the menace of Russia. The style is factual, lacking descriptive interest, and somewhat monotonous since it makes no great attempt to outline the larger picture of the war against the personal and necessarily narrower viewpoint of the author as a fighter pilot.

His courage and devotion one accepts. He made 400 operational sorties and logged 2,000 hours after making an unspectacular start with his tuition. What particular quality other than tenacity made him so good a fighter is not clear from his account, yet he was credited with 52 hostile aircraft. The story of these victories is somewhat overloaded with such phrases as: "I resolve with grim determination."—P.

SPACE MEDICINE. *Edited by John P. Marbarger. The University of Illinois Press 1951. 83 pp. $3.00 net.*

As indicated by Major General Armstrong in the first chapter of this small book, *Space Medicine* is a theoretical exercise into the physical and physiological factors that are likely to be encountered in travel beyond the stratosphere. Apart from one chapter on the physiological considerations of the possibility of life under extra-terrestrial conditions, the rest of the book is devoted to the physical, astronomical and biochemicalogical aspect of projected travel beyond the earth. The book makes interesting reading and may be calculated to stimulate the day dreams or nightmares of those interested in this subject.

Although well presented with clear diagrams and printed on good paper it is doubtful whether at 3d. a page this small volume would have much demand in this country.—B-B.

Additions to the Library

Acoustical Society of America. SYMPOSIUM ON AIRCRAFT NOISE. Acoustical Society of America. 1953.
Cornell Aeronautical Laboratory. AN EXPERIMENTAL AND THEORETICAL STUDY OF THE LOW FREQUENCY FLAPPING-PITCHING FLUTTER OF A HOVERING ROTOR BLADE. Cornell Aeronautical Laboratory. 1952.
Cunningham, A. TUMULT IN THE CLOUDS. Peter Davies. 1953.
Gibbs-Smith, C. H. A HISTORY OF FLYING. Batsford. 1953.
1952 HEAT TRANSFER AND FLUID MECHANICS INSTITUTE PREPRINTS. Stanford University. 1952.
Hensser, H. COMET HIGHWAY. John Murray. 1953.
Jack, A. FEATHERED WINGS. Methuen. 1953.
Johnson, L. H. NOMOGRAPHY AND EMPIRICAL EQUATIONS. John Wiley. 1952.
Materials Handling Sub-Committee. REVIEW OF MATERIALS HANDLING IN BRITISH MANUFACTURING INDUSTRIES. Institution of Production Engineers. 1953.
Moore, H. F. and Moore, M. B. TEXTBOOK OF THE MATERIALS OF ENGINEERING. 8th edition. McGraw Hill. 1953.
Pars, L. A. INTRODUCTION TO DYNAMICS. Cambridge University Press. 1953.
Petter, W. E. W. DESIGN FOR PRODUCTION. Folland Aircraft. 1953.
Reynolds, Q. THE AMAZING MR. DOOLITTLE. Appleton-Century-Crofts. 1953.
Strughold, H. THE GREEN AND RED PLANET. University of New Mexico. 1953.

Reports

AERODYNAMICS

Boundary Layer

Wind-tunnel tests on the 30 per cent. symmetrical Griffith aerofoil with distributed suction over the nose. N. Gregory, W. S. Walker and A. N. Devereux. R. & M. 2647. (June 1948, published 1953.)

This report describes tests made on the 30 per cent. Griffith symmetrical aerofoil with continuous suction applied through a porous capping fitted over the front 15 per cent. of the upper surface. The profile drag of the aerofoil was measured, and estimates were made of the equivalent drag coefficients for the work done by the suction pumps. Measurements of the velocity within the boundary layer were made at various chordwise positions on the porous surface; the profiles recorded were very close to the theoretical.—(1.1.5).

An experimental investigation of the influence of strong adverse pressure gradients on turbulent boundary layers at supersonic speeds. G. Drougge. F.F.A., Sweden. Report N.R 46 (1953).

The investigation deals mainly with the influence on turbulent boundary layers of a shock wave, originating at a concave corner. For the sake of completeness and comparison a few measurements on laminar boundary layers were made and also the influence of a continuous compression was investigated.—(1.1.3 × 1.2.3).

Compressible Flow

See also Internal Flow, Boundary Layer and Reference Literature

The design and testing of supersonic nozzles. R. Harrop, P. I. F. Bright, J. Salmon and M. T. Cainger. R. & M. 2712. (May 1950, published 1953.)

The theory of the flow through a throat near sonic velocity is developed, and is followed by a discussion of the conventional method of designing supersonic nozzles using the method of characteristics. A method of improving the Mach number distribution of the nozzle using the experimental results is developed. The nozzles designed were tested in a 3-in. square wind tunnel in which the Mach number distribution was obtained by shaping the top wall of the working-section. The flow through the throat of the liners was determined experimentally and compared with the theory. The agreement was good on the whole, although there were differences in the subsonic entry region because the bottom wall only became flat a short distance before the throat. In addition, tests were made which showed that the assumption of two-dimensional flow through the throat was justified.—(1.2.3).

Theoretical pressure distributions and wave drags for conical boat tails. J. R. Jack. N.A.C.A. Technical Note 2972. (July 1953.)

After-body pressure distributions and wave drags were calculated using a second-order theory for a variety of conical boat tails at zero angle of attack. Results are presented for Mach numbers from 1·5 to 4·5, area ratios from 0·200 to 0·800, and boat tail angles from 3° to 11°.—(1.2.3).

Fluid Dynamics

Air-Water analogy and the study of hydraulic models. G. Supino. N.A.C.A. Technical Memorandum. 1359. (July, 1953.)

The author first sets forth some observations about the theory of models. Then he establishes certain general criteria for the construction of dynamically similar models in water and in air, through reference to the perfect fluid equations and to those pertaining to viscous flow.—(1.4).

Internal Flow

Corrections to velocity for wall constraint in any 10 × 7 rectangular subsonic wind tunnel. J. Y. G. Evans. R. & M. 2662. (April 1949, published 1953.)

The validity and accuracy of methods of determining corrections to the measured velocity in a wind tunnel to compensate for the constraining effect of the walls are reviewed following recent experimental evidence from the R.A.E. 10 × 7 ft. subsonic wind tunnel. It is concluded that such corrections, commonly known as "blockage" corrections, can be successfully applied at Mach numbers up to 0·96 but some modifications are necessary to the formulæ at present in use.—(1.5.1).

Some high-speed tests on turbine cascades. E. A. Bridle. R. & M. 2697. (February 1949, published 1953.)

High-speed wind-tunnel tests on seven cascades of turbine blades are described, the blades having conventional sections including both reaction and impulse designs. The two-dimensional performance over wide ranges of incidence at Mach numbers up to 1·0 is discussed, special importance being attached to the effects of compressibility.—(1.5.4).

The improvement in pressure recovery in supersonic wind tunnels. H. Eggink. R. & M. 2703. (May 1949, published 1953.)

The inefficient pressure recovery of present day supersonic wind tunnels, which leads to high costs of plant installation and operation, is discussed and methods of improvement suggested. In particular, the diffuser system, where most of the losses occur, is studied in detail; the improvement to be expected in the pressure recovery by the use of convergent-divergent types is explained and methods of overcoming the necessity for high starting powers with this arrangement are presented.—(1.5.1).

Experiments on conical diffusers. H. B. Squire. R. & M. 2751. (November 1950, published 1953.)

Measurements of the characteristics of conical diffusers of area ratio 4 between entry and exit and of total cone angle 4, 5, 6, 8 and 10 deg. were made to determine the effect of Reynolds number on efficiency for uniform entry flow. A range of entry Reynolds number between 5×10^4 and 10^6 was covered.—(1.5.1).

The theoretical pressure distributions around some conventional turbine blades in cascade. T. J. Hargest. R. & M. 2765. (March 1950, published 1953.)

By means of Relf's analogy between aerodynamic streamline flow and electric potential flow, the theoretical pressure distributions around a series of conventional turbine blades in cascade have been determined over a range of incidence covered in some previously reported aerodynamic tests. The theoretical pressure distributions and their variation with incidence provide the basis of an explanation of the observed aerodynamic performance.—(1.5.4).

Effect of blade-thickness taper on axial-velocity distribution at the leading edge of an entrance rotor-blade row with axial inlet and the influence of this distribution on alinement of the rotor blade for zero angle of attack. John D. Stanitz. N.A.C.A. Technical Note 2986. (August 1953.)

A method is developed for estimating the effect of blade thickness taper on the inlet axial-velocity distribution of an entrance rotor-blade row with axial inlet, and the influence of this velocity distribution on the alignment of the rotor blade for zero effective angle of attack. This alignment of the blade requires a deviation between the angle of the blade camber line direction at the inlet and the upstream relative flow direction.—(1.5.4).

NOTE:—*The figures in parenthesis at the end of each Summary are for office use only.*

Comparison of secondary flows and boundary-layer accumulations in several turbine nozzles. Milton G. Kofskey, Hubert W. Allen and Howard Z. Herzig. N.A.C.A. Technical Note 2989. (*August* 1953.)

An investigation was made of losses and secondary flows in three different turbine nozzle configurations in annular cascades. Appreciable outer shroud loss cores (passage vortices) were found to exist at the discharge of blades which had thickened suction surface boundary layers near the outer shroud. Blade designs having thinner boundary layers did not show such outer shroud loss cores, but indicated greater inward radial flow of low momentum air in the blade wake which resulted in greater contribution to inner shroud loss regions.—(1.5.4).

Application of a characteristic blade-to-blade solution to flow in a supersonic rotor with varying stream-filament thickness. Eleanor L. Costilow. N.A.C.A. Technical Note 2992. (*August* 1953.)

An analysis of the circumferential blade-to-blade flow properties in a supersonic impeller has been made using the method of characteristics on an arbitrary stream surface of revolution. The method takes into account variable stream-filament thickness and curvature along the flow surface.—(1.5.2 × 1.2.3)

LOADS

See also STRUCTURES, LOADS

*Influence of compressibility on the calculated flexure-torsion flutter speed of a family of rectangular cantilever wings. J. Iff. N.L.L., Holland. Report F.*118.

The flutter speed of 27 wings with different positions of elastic and inertia axes and different values of the relative density parameter has been calculated for three values of the Mach number. It has been found that the influence of compressibility can be as well favourable as unfavourable, depending upon the case investigated. All results are presented in a number of diagrams which also show the influence of the other parameters on the flutter speed.—(1.6).

PERFORMANCE ESTIMATION

*Tests on the Hurricane L.*1696 *in the* 24 *ft. wind tunnel. D. W. Bottle and T. V. Somerville. R. & M.* 2562. (*August* 1941, *published* 1953.)—(1.7).

STABILITY AND CONTROL

Definitions of the angles of incidence and of sideslip. C. H. E. Warren. A.R.C. Current Paper No. 124. (*August* 1952, *published* 1953.)

The use of large angles of incidence and of sideslip in missile work, and recent changes in wind tunnel testing techniques, have shown the need for clear and precise definitions of the angles of incidence and of sideslip. The suitability of different definitions for both experimental and theoretical work in both the aircraft and missile field is considered, and it is concluded that as no single definition is universally acceptable, care should be taken in theoretical and experimental reports to define precisely the angles used.—(1.8).

*An analysis of the steady climbing turn without sideslip. J. R. Baxter. A.R.L., Australia. Report A.*81. (*December* 1952.)

The geometry of the system and the equilibrium of the principal forces have been completely analysed without approximations, and the six equations of motion have been set out in their accurate dimensional form. Then, introducing a series of approximations consistent with those in Report A.76 (The analysis of the idealised pull-out in a vertical plane), the various equations have been simplified and solved. The solutions have been used to derive expressions for the two manœuvrability criteria, stick movement and stick force per *g*, and comparisons have been made with the equivalent criteria for the pull-out case.—(1.8).

Contribution a l'étude expérimentale des mouvements variés en soufflerie. M. Scherer. O.N.E.R.A., France. Publication No. 61. (1953.)—(1.8 × 2.0).

Sur la stabilité des avions. J.-M. Souriau. O.N.E.R.A., France. Publication No. 62. (1953.)—(1.8).

Analysis of some wing-body-vertical tail interference problems for non-symmetric steady flow using slender-body theory. M. T. Landahl. K.T.H., Sweden. Aero T.N. 32. (*July* 1953.)

The steady flow about certain wing-body-tail configurations is studied by using the slender-body theory. The type of configuration consists of a wing and an asymmetrically placed tail of infinitesimal thickness mounted on a body of revolution. In the main part of the work, the flow at sideslip and incidence is studied. In addition, other wake interference problems are studied. Thus the influence of discrete vortices on the configuration and the lift on an after-body extending behind the trailing edge are calculated. The analysis is made in such a way that the vorticity distribution in the wake may be put in afterwards.—(1.8).

A flight investigation of the effect of steady rolling on the natural frequencies of a body-tail combination. Norman R. Bergrun and Paul A. Nickel. N.A.C.A. Technical Note 2985. (*August* 1953.)

Flight data have been obtained with a freely falling body-tail combination to show the effects of steady rolling on the response in the pitching and yawing motions. Observed effects are compared with those predicted by theory, and reasonably good agreement is obtained. Charts are also presented to provide a rapid means for estimating the effective values of the stability derivatives C_{m_α} and C_{n_β} under steady rolling conditions.—(1.8).

WINGS AND AEROFOILS

Surface slopes and curvatures of the R.A.E. 100-104 *and other rooftop-type aerofoil sections. J. Williams and Edna M. Love. A.R.C. Current Paper No.* 129. (*October* 1952, *published* 1953.)

Formulæ and tables have been obtained for the accurate and rapid calculation of the surface slopes and curvatures of the R.A.E. 100-104 aerofoil sections, and of more general "roof top-type" sections. In particular, the surface slopes of the R.A.E. 102 and 104 shapes have been evaluated for application with a "tangent-plane" method of model construction.—(1.10).

A calculation of the complete downwash in three dimensions due to a rectangular vortex. Doris E. Lehrian. R. & M. 2771. (*March* 1949, *published* 1953.)

A calculation of the complete downwash in three dimensions due to a rectangular vortex is given for the limited range $Z = \pm 4$. The downwash is computed at *s* selected positions, in planes normal to the plane of the vortex; these planes are spaced at even integral multiples of the semi-width of the vortex, measured from the line of symmetry. Values are tabulated for *Z* in the range (0·4) and a set of graphs is also included for $0 \leq Z \leq 2$; they are to be used in conjunction with the "Tables of Complete Downwash due to a Rectangular Vortex" (R. & M. 2461).—(1.10.1.2).

On the subsonic, transonic and supersonic flow around low aspect ratio wings with incidence and thickness. F. Keune. K.T.H., Sweden. Aero T.N. 28. (*September* 1953.)

The flow on the actual wing surface is calculated for the case of a wing with incidence and thickness by extending R. T. Jones' theory (1) for the wing with incidence but with zero thickness, and F. Keune's theory (2) for the wing with thickness at zero incidence. In addition to these theories one obtains contributions to velocity, pressure distribution, lift and moment coefficients, which are determined only by the two-dimensional cross-sectional flow.—(1.10.1.2).

The flat triangular wing with subsonic leading edges in sideslip at supersonic speeds. H. Behrbohm. S.A.A.B., Sweden. T.N. 14. (*October* 1952.)

The sideslipping triangular wing with subsonic leading edges at supersonic speeds is treated in quite an elementary manner. Through a hyperbolic rotation the flow problem of the symmetrical flat triangle at an angle of sideslip is transformed into the well known problem of the triangular wing with no sideslip but with oblique (but supersonic) trailing edge. Without complex mathematics the pertubation velocity potential and the velocity pertubations (including

the upwash in front of the wing) are obtained both in windfixed and in bodyfixed co-ordinates. The spanwise load distribution can be given in a simple closed form from which total lift and rolling moment due to sideslip are easily deduced. Sideforce and yawing moment are also given in the whole considered range of the angle of slideslip.—(1.10.1.2).

Transonic flow past a wedge profile with detached bow wave. Walter G. Vincenti and Cleo B. Wagoner. N.A.C.A. Report 1095. (1952.)

A theoretical study has been made of the aerodynamic characteristics at zero angle of attack of a thin, doubly symmetrical double-wedge profile in the range of supersonic flight speed in which the bow wave is detached. The analysis utilises the equations of the transonic small-disturbance theory and involves no assumptions beyond those implicit in this theory. The mixed flow about the front half of the profile is calculated by relaxation solution of a boundary-value problem for the transonic small-disturbance equation in the hodograph plane. The methods follow established lines except for the somewhat novel treatment of the boundary conditions along the shock polar and sonic line. The purely supersonic flow about the rear of the profile is found by means of the method of characteristics specialised to the transonic small-disturbance theory.—(1.10.1.1).

HELICOPTER AERODYNAMICS

Dynamic longitudinal stability measurements on a single-rotor helicopter (Hoverfly Mk. I). W. Stewart. R. & M. 2505. (*February* 1948, *published* 1953.)

Flight measurements have been made of the phugoid motion of the Hoverfly Mk. I helicopter, following an arbitrary longitudinal displacement of the control, the latter being returned to its initial position and held fixed. The tests were done throughout the speed range for power-on conditions and in autorotation for various centre of gravity positions and for forward and backward initial displacement of the stick.—(1.11).

Wind-tunnel tests on a 12 ft. diameter helicopter rotor. H. B. Squire, R. A. Fail and R. C. W. Eyre. R. & M. 2695. (*April* 1949, *published* 1953.)

Measurements of the thrust, torque and flapping angle for a 12-ft. diameter rotor over a range of blade angle, shaft inclination and tip-speed ratio have been made to give information on the validity of the standard rotor theory and of the effect of stalling on the retreating blade.—(1.11).

AEROELASTICITY

Aerofoil oscillations at high mean incidences. W. P. Jones. R. & M. 2654. (*April* 1948, *published* 1953.)

The problem of the estimation of the aerodynamic forces acting on two-dimensional aerofoils oscillating at mean incidences below the stall is considered. A method of calculation is suggested which makes use of the steady motion characteristics of the aerofoil. At low frequencies, good agreement with the measured aerodynamic derivatives should be obtained as the method is such that it gives the correct values at zero frequency. A comparison between the estimated and measured values of the pitching-moment derivatives for a particular aerofoil is made, and this shows that the method suggested gives better agreement with experiment than the usual vortex-sheet theory.—(2).

AIRCRAFT DESIGN AND CONSTRUCTION

Landing of an aircraft on a suspended sheet. J. Taylor. R. & M. 2574. (*June* 1947, *published* 1953.)

An investigation is made into the characteristics of a freely suspended flexible sheet as a shock absorber replacing the conventional undercarriage, particular attention being given to the inertia of the sheet. The effect of the aircraft having a forward component of velocity increases the contribution of sheet momentum. For reasonably practical landing speeds and sheet dimensions, virtually the whole of the momentum of descent is absorbed by sheet inertia. Under such conditions still higher retardation efficiencies are obtainable and, with a suitable design of aircraft keel, rebound may be entirely eliminated.—(4.2).

Studies of the lateral-directional flying qualities of a tandem helicopter in forward flight. Kenneth B. Amer and Robert J. Tapscott. N.A.C.A. Technical Note 2984. (*August* 1953.)

Results of flight measurements and pilots' opinions of the forward-flight lateral-directional flying qualities of a tandem helicopter in several different configurations are compared and analysed. The conclusions are presented in the form of desirable flying-quality goals applicable to all types of helicopters.—(4.4).

Flight measurements and analysis of helicopter normal load factors in maneuvers. F. B. Gustafson and Almer D. Crim. N.A.C.A. Technical Note 2990. (*August* 1953.)

Flight tests have been conducted in two single-rotor helicopters to determine the load factors due to various manœuvres. A simple theoretical method of estimating for new designs the maximum load factor that can be attained for any given flight condition is also presented, and the results compared with those obtained from the flight tests.—(4.4).

AIRCRAFT OPERATION

A method for calculating the evaporation from water sprays in an icing tunnel. E. L. Smith and O. R. Ballard. N.A.E., Canada, Laboratory Report LR-60. (*May* 1953.)—(5.4).

FLIGHT TESTING

Pressure error measurement using the formation method. K. C. Levon. A.R.C. Current Paper No. 126. (1953.)

Measurements of pressure error at altitude have been made by flying several aircraft in formation with a reference aircraft whose air speed system had previously been calibrated by radar.—(13.1).

FUELS AND LUBRICANTS

Correlation of physical properties with molecular structure for some dicyclic hydrocarbons having high thermal-energy release per unit volume—2-alkylbiphenyl and the two isomeric 2-alkylbicyclohexyl series. I. A. Goodman and P. H. Wise. N.A.C.A. Report 1065. (1952.)

Three homologous series of related dicyclic hydrocarbons are compared on the basis of the following physical properties.: net heat of combustion, density, melting point, boiling point, and kinematic viscosity. The three series investigated are the 2-alkylbiphenyl, and the high- and low-boiling 2-alkylbicyclohexyl series through C_{16}.—(14.1).

HYDRODYNAMICS

Nouveaux compléments d'hydraulique. Première partie. L. Escande. Publications Scientifiques et Techniques du Ministère de L'Air, France. No. 280. (1953.)—(17.1).

The high-speed planing characteristics of a rectangular flat plate over a wide range of trim and wetted length. Irving Weinstein and Walter J. Kapryan. N.A.C.A. Technical Note 2981. (*July* 1953.)

The principal high-speed planing characteristics for a prismatic surface having an angle of dead rise of 0° (flat bottom) have been determined over a wide range of planing variables. Wetted length, resistance, centre-of-pressure location, and draft were determined at speed coefficients ranging up to 25·0, beam loadings up to 87·3, and trims up to 30°. Mean wetted lengths up to 7·0 beams were obtained wherever possible.—(17.2).

INSTRUMENTS AND EQUIPMENT

Techniques for calculating parameters of nonlinear dynamic systems from response data. Benjamin R. Briggs and Arthur L. Jones. N.A.C.A. Technical Note 2977. (*July* 1953.)

A study has been made of the problem of determining parameters of dynamic systems, such as aircraft or servo-mechanisms, whose measured responses are non-linear. The parameters studied were functions of displacement of velocity.—(18).

MATERIALS

Properties of Metallic Materials

See also Structures, Structural Testing

Tensile and endurance tests on preformed stainless steel cable. A. G. Cole, R. A. Fell and W. W. Johnstone. A.R.L. Australia. Structures and Materials Note 206. (*June* 1953.)

To investigate failures in service, specification tensile and endurance tests were carried out on preformed stainless steel cable. For comparison tests were also carried out on cables obtained from three other manufacturers. The investigation showed that none of the cables supplied complied fully with the specification requirements.—(21.2).

Direct fatigue tests with tensile and compressive mean stresses on 24S-*T aluminium plain specimens and specimens notched by a drilled hole. G. Wallgren. FFA Sweden. Report NR* 48. (1953.)

The determination of the direct fatigue strength of materials and structural elements has hitherto mainly been limited to stress ranges where the mean stress was either zero or tensile. This investigation has been extended to cover fatigue tests for both positive and negative values of the mean stress. The purpose is to study how the fatigue strength depends on the magnitude of this stress and tests have been made with polished specimens of aluminium alloy 24S-T both with and without notches. The specimens were of circular cross section and the notch consisted of a transverse hole.—(21.2).

Structural efficiencies of various aluminum, titanium, and steel alloys at elevated temperatures. George J. Heimerl and Philip J. Hughes. N.A.C.A. Technical Note 2975. (*July* 1953.)

Efficient temperature ranges are indicated for two high-strength aluminium alloys, two titanium alloys, and three steels for some short-time compression-loading applications at elevated temperatures. Only the effects of constant temperatures and short exposure to temperature are considered, and creep is assumed not to be a factor. The structural efficiency analysis is based upon preliminary results of short-time elevated-temperature compressive stress-strain tests of the materials. The analysis covers strength under uniaxial compression, elastic stiffness, column buckling, and the buckling of long plates in compression or in shear.—(21.2 × 33).

MATHEMATICS

Some related oscillation problems. W. J. Duncan. R. & M. 2707. (*April* 1949, *published* 1953.)

Two simple means for establishing a relation between a pair of oscillation problems are discussed briefly. In the first, the displacements are connected by use of a differential operator. The set of natural frequencies is identical for the two problems and results of interest are obtained when the transformed boundary conditions can be physically interpreted. In this manner it is shown, for example, that a flywheel on a uniform shaft can be transformed into a flexible coupling and a mass carried on a uniform beam into a flexible hinge. In the second, the connection is established by use of the concept of mechanical admittance. Here the frequency equations are simply related but the frequencies are not.—(22).

Actes du Colloque International des Vibrations non Linéaires. Ile de Porquerolles 1951. *Publications Scientifiques et Techniques du Ministère de L'Air, France. No.* 281. (1953.)—(22).

Régularité et séries divergentes. P. Vernotte. Publications Scientifiques et Techniques du Ministère de L'Air, France. No. 282. (1953.)—(22).

REFERENCE LITERATURE

A revised index of mathematical tables for compressible flow. R. C. Tomlinson. R. & M. 2691. (*December* 1949, *published* 1953.)—(30.1 × 1.2).

RESEARCH

An electrical analogy for solving the oscillating-surface problem for incompressible nonviscid flow. M. T. Landahl and V. J. E. Stark. KTH Sweden, Aero TN 34. (*July* 1953.)

A method is suggested for solving the problem of an oscillating surface in three-dimensional, incompressible, ideal flow with the aid of an electric potential tank. A special potential function, satisfying the Laplace equation, is used and the boundary conditions are formulated in this potential.—(31.1).

STRUCTURES

Loads

*Application of experimental aerodynamic coefficients to flutter calculations. H. Bergh and J. Ijff. N.L.L. Holland. Report F.*122. (*November* 1952, *published* 1953.)

Experimental aerodynamic coefficients of an oscillating wing in two-dimensional flow have been used in bending-torsion flutter calculations of several rectangular wings. The experimental aerodynamic coefficients are given in the form of graphs and tables for values of the reduced frequency ω from 0·08 to 1.—(33.1 × 1.6).

An analysis of normal-acceleration and airspeed data from a four-engine type of transport airplane in commercial operation on an Eastern United States route from November 1947 *to February* 1950. *Thomas L. Coleman and Paul W. J. Schumacher. N.A.C.A. Technical Note* 2965. (*August* 1953.)

Normal acceleration and air speed data covering about 50,000 hours of flight of one type of four-engine transport aeroplane flown in commercial operations at altitudes to 20,000 feet on a north-south route in the eastern United States are analysed to determine the gusts encountered and the accelerations experienced.—(33.1.2).

Calibration of strain-gage installations in aircraft structures for the measurement of flight loads. T. H. Skopinski, William S. Aiken, Jr. and Wilber B. Huston. N.A.C.A. Technical Note 2993. (*August* 1953.)

A general procedure is developed for calibrating strain-gauge installations in aircraft structures for application to flight measurements of loads. The basic procedure can be modified as necessary to suit the requirements of any particular structure. The application of the procedure is illustrated by results for two typical structures.—(33.1).

Theory and Analysis

Elasticity of a sheet reinforced by stringers and skew ribs, with applications to swept wings. E. H. Mansfield. R. & M. 2758. (*December* 1949, *published* 1953.)

A rigorous theory has been developed for determining the stresses and displacements in a sheet reinforced by stringers and ribs which are not at right-angles to the stringers. The solution of many problems of practical importance has been facilitated by the introduction of a stress function. The theory has been applied to a cylinder of rectangular section stiffened with such skew ribs (a simplified representation of a swept wing). It is shown that there are axes about which applied moments produce pure twist or pure curvature of the cylinder. There are simple formulæ for determining these axes and the relationships between twist and curvature and the applied moments.—(33.2).

Stresses in built-up beams due to an abrupt change in shear stress at a loading station. J. Taylor. R. & M. 2775 (*August* 1949, *published* 1953.)

Owing to the abrupt change in shear stress at loading sections of beams there is a concentration of direct stress in the outer fibres of the beam near the loading section. A method of calculating this concentration is described. The highest stress concentrations occur in short deep beams and are greater for wooden than metal beams. The method is applied to the spars of two wooden aircraft and stress concentrations 1·06 and 1·4 are found at the fuselage attachments. Strain measurements were made at positions on a wooden beam under load and the theoretical predictions verified.—(33.2).

The statistical significance of a few fatigue results. G. J. Schott. N.A.E. Canada. Laboratory Report LR-58. (May 1953.)

This report describes how statistical predictions can be made regarding the fatigue life of a structure, if the fatigue test results of only a few nominally similar specimens are available. The detailed numerical procedure involved is described.—(33.2 × 21.2).

STRUCTURAL TESTING

Analysis of experiments on swept wing structures. D. Howe. College of Aeronautics, Cranfield. Report No. 65. (May 1953.)

The following work is considered:—A single-cell swept-wing with ribs normal to the spars, a strain energy solution, allowing for a flexible root rib and shear lag, being applied. A two-cell swept wing, with ribs normal to the spars, again using a strain energy solution. A two-cell swept wing with ribs parallel to the line of flight, where oblique co-ordinate theory is used. A single-cell swept box, having oblique ribs, where oblique co-ordinate theory is found to give good results away from the root. A comparison of wings with ribs normal to the spars and parallel to the line of flight is made.—(33.3).

Some preliminary model experiments on the whirling of shafts. E. Downham. R. & M. 2768. (June 1950, published 1953.)

Experimental work is now being done to establish a basis for the solution of whirling problems on turbine and contra-rotating shaft systems in the design stage. This report is concerned primarily with the degree of accuracy to be expected from experiments on models. In experiments here described results are obtained for a simple cantilever system which are in close agreement with theory. With more complicated systems the error is somewhat greater owing to practical effects not covered by theory, although still acceptable for most design purposes.—(33.3).

Experimental investigation of the stress distribution in a plastic model of a 35° swept back wing with multi-web construction. B. R. Noton. FFA Sweden. Report NR 47. (1953.)

This report covers the experimental determination of the stress patterns in the webs and over the top surface on a model of a 35° swept-back wing, with five spars and a few ribs placed perpendicular to the rear spar. The model included the structure located inside the fuselage of the aircraft and was supported at 4 couplings, representing the fuselage connections. The material of the model was plastic (methyl methacrylate) and the major dimensions were to a scale of one-fifth.—(33.3).

WEIGHT ANALYSIS AND CONTROL

Comparison of the Aeronautical Research Laboratories and Royal Aircraft Establishment methods of wing weight prediction. J. Solvey. A.R.L. Structures and Materials Technical Memorandum 31 (June 1951).

The advantages of the Aeronautical Research Laboratories method of wing weight prediction are indicated, as well as its value in the design and assessment of new wing structures.—(33.4.1).

Determination of optimum dimensions for an aircraft fuselage shell considering special stiffness criteria and minimum weight. P. H. B. Schalin. S.A.A.B. T.N. 8 (14 July 1951).

The problem of distributing the bending and torsional material in aircraft main structures for minimum weight with regard to stiffness considerations is presented. A solution based upon special criteria for bending and torsional deflections is given for the part of the fuselage between wing and tail. Numerical evaluations show a substantial weight saving made possible for the configuration in question.—(33.4.1).

THERMODYNAMICS

THERMODYNAMIC PROPERTIES

Structure ae l'onde de choc et des flammes déflagrantes. M. Roy. O.N.E.R.A. Publication No. 63. (1953.)—(34.1.1).

THERMAL MEASUREMENT AND CONTROL

The condensation line of air and the heats of vaporization of oxygen and nitrogen. G. T. Furukawa and R. E. McCoskey. N.A.C.A. Technical Note 2969 (June 1953).

The condensation pressure of air was determined over the range of temperature from 60° to 85° K. The experimental results were slightly higher than the calculated values based on the ideal solution law. Heat of vaporisation of oxygen was determined at four temperatures ranging from about 68° to 91° K., and of nitrogen at four temperatures ranging from 62° to 78° K.—(34.2.2).

HEAT TRANSFER

The determination of skin temperatures attained in high speed flight. F. V. Davies and R. J. Monaghan. A.R.C. Current Paper No. 123. (February 1952, published 1953.)

The factors affecting skin temperatures attained by high speed missiles are discussed and some methods of solution are presented. These have been reduced to graphical or tabular form and are set out in order of complexity.—(34.3).

Note on the effect of variable wall temperature on heat transfer. H. B. Squire. R. & M. 2753. (October 1950, published 1953.)

This report is an addendum to "Heat Transfer Calculation for Aerofoils" by H. B. Squire (R. & M. 1986, November 1942).—(34.3).

A new method for calculating laminar heat transfer on cylinders of arbitrary cross-section and on bodies of revolution at constant and variable wall temperature. H. Schuh. K.T.H. Aero. T.N. 33 (June 1953).

A method of calculation is developed for flow with constant fluid properties and without heat produced by friction: the neglected effects can be taken into account later on by suitably chosen reference temperatures. The same method is applicable to boundary layers on cylinders and on bodies of revolution. Among examples tested are a flat plate at zero incidence, two elliptic cylinders and a body of revolution: in these cases "overall" and "partial" heating of the body has been calculated: with partial heating, heat transfer begins some distance downstream of the stagnation point.—(34.3.2).

Experiments on mixed-free-and-forced-convective heat transfer connected with turbulent flow through a short tube. E. R. G. Eckert, et al. N.A.C.A. Technical Note 2974. (July 1953.)

Convective heat-transfer experiments were conducted for turbulent, mixed-free- and -forced-convection flow through a vertical heated tube with a length-to-diameter ratio of 5. Studies were made with forced air flowing parallel or opposite to the free convection flow. The Grashof numbers ranged from 10^9 to 10^{13} and the Reynolds numbers, from 36×10^3 to 377×10^3.—(34.3.2).

TRAINING

The instrumentation of an aircraft (de Havilland "Dove") for flight test instruction. C. F. Bethwaite and A. F. Newell. College of Aeronautics Report No. 70. (March 1953.)—(35).

APPOINTMENTS

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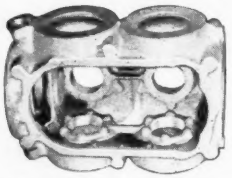

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